NASA CR-132575-3

SUBSTANTIATING DATA FOR

ARROW-WING SUPERSONIC CRUISE AIRCRAFT STRUCTURAL DESIGN CONCEPTS EVALUATION

Volume 3, Sections 12 through 14

(NASA-CR-132575-3) ARROW-WING SUPERSONIC CRUISE AIRCRAFT STRUCTURAL DESIGN CONCEPTS EVALUATION. VOLUME 3: SECTIONS 12 THROUGH 14 (Lockheed-California Co.) 330 p HC $10.00 CSCL 01C G3/05
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by
I. F. Sakata and G. W. Davis

Prepared under Contract No. NAS1-12288
Lockheed-California Company
Burbank, California

for Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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SUMMARY

An analytical study was performed to determine the structural approach best suited for the design of a Mach 2.7 arrow-wing supersonic cruise aircraft.

Results, procedures, and principal justification of results are presented in Reference 1. Detailed substantiation data are given herein. In general, each major analysis is presented sequentially in separate sections to provide continuity in the flow of the design concepts analysis effort. In addition to the design concepts evaluation and the detailed engineering design analyses, supporting tasks encompassing: (1) the controls system development (2) the propulsion-airframe integration study, and (3) the advanced technology assessment are presented.

Reference 1 Sakata, I. F. and Davis, G. W.: Evaluation of Structural Design Concepts for an Arrow-Wing Supersonic Cruise Aircraft NASA CR- 1976

INTRODUCTION

The design of an economically viable supersonic cruise aircraft requires reduced structural mass fractions attainable through application of new materials, advanced concepts and design tools. Configurations, such as the arrow-wing, show promise from the aerodynamic standpoint; however, detailed structural design studies are needed to determine the feasibility of constructing this type of aircraft with sufficiently low structural mass fraction.

For the past several years, the NASA Langley Research Center has been pursuing a supersonic cruise aircraft research program (1) to provide an expanded technology base for future supersonic aircraft, (2) to provide the data needed to assess the environmental and economic impacts on the United States of present and especially future foreign supersonic cruise aircraft, and (3) to provide a sound technical basis for any future consideration that may be given by the United States to the development of an environmentally acceptable and economically viable commercial supersonic cruise aircraft.

The analytical study, reported herein, was performed to provide data to support the selection of the best structural concept for the design of a supersonic cruise aircraft wing and fuselage primary structure considering near-term start-of-design technology. A spectrum of structural approaches for primary structure design that has found application or had been proposed for supersonic aircraft design; such as the Anglo-French Concorde supersonic transport, the Mach 3.0-plus Lockheed F-12 and the proposed Lockheed L-2000 and Boeing B-2707 supersonic transports were systematically evaluated for the given configuration and environmental criteria.

The study objectives were achieved through a systematic program involving the interactions between the various disciplines as shown in Figures A through C. These figures present an overview of the study effort and provides a summary statement of work, as follows:

(1) Task I - Analytical Design Studies (Figure A).- This initial task involved a study wherein a large number of candidate structure

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concepts were investigated and subjected to a systematic evaluation process to determine the most promising concepts. An airplane configuration refinement investigation, including propulsion-airframe integration study were concurrently performed.

(2) Task II - Engineering Design/Analyses (Figure B).- The most promising concepts were analyzed assuming near-term start-of-design technology, critical design conditions and requirements identified, and construction details and mass estimates determined for the Final Design airplane. Concurrently, the impact of advanced technology on supersonic cruise aircraft design was explored.

(3) Task III - Mass Sensitivity Studies (Figure C).- Starting with the Final Design airplane numerous sensitivity studies were performed. The results of these investigations and the design studies (Task I and Task II) identified opportunities for structural mass reduction and needed research and technology to achieve the objectives of reduced structural mass.

Displayed on the figures are the time-sequence and flow of data between disciplines and the reason for the make-up of the series of sections presented in this report. The various sections are independent of each other, except as specifically noted. Results of this structural evaluation are reported in Reference 1. This reference also includes the procedures and principal justification of results, whereas this report gives detailed substantiation of the results in Reference 1. This report is bound as four separate volumes.

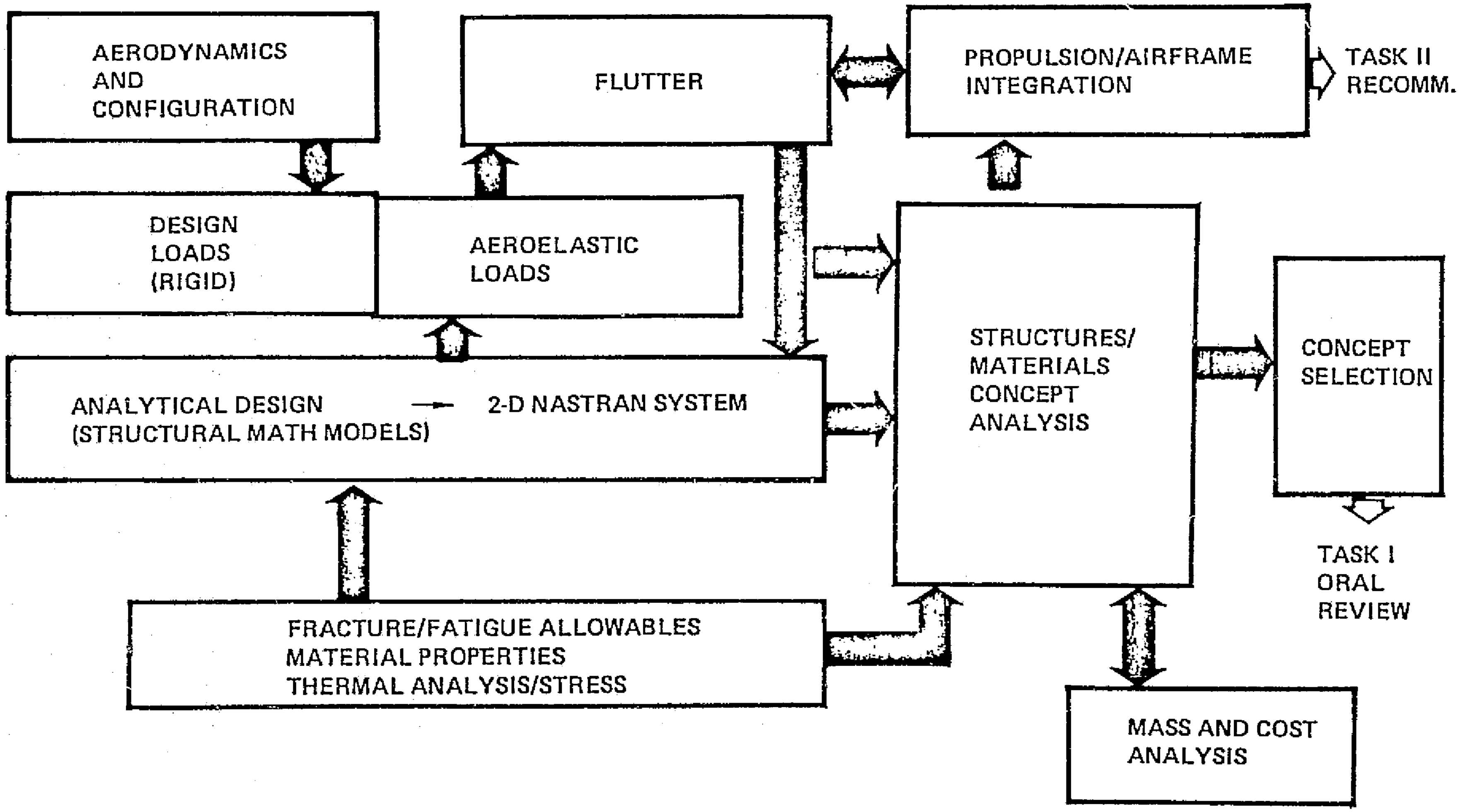

Figure A. Analytical Design Studies - Task I

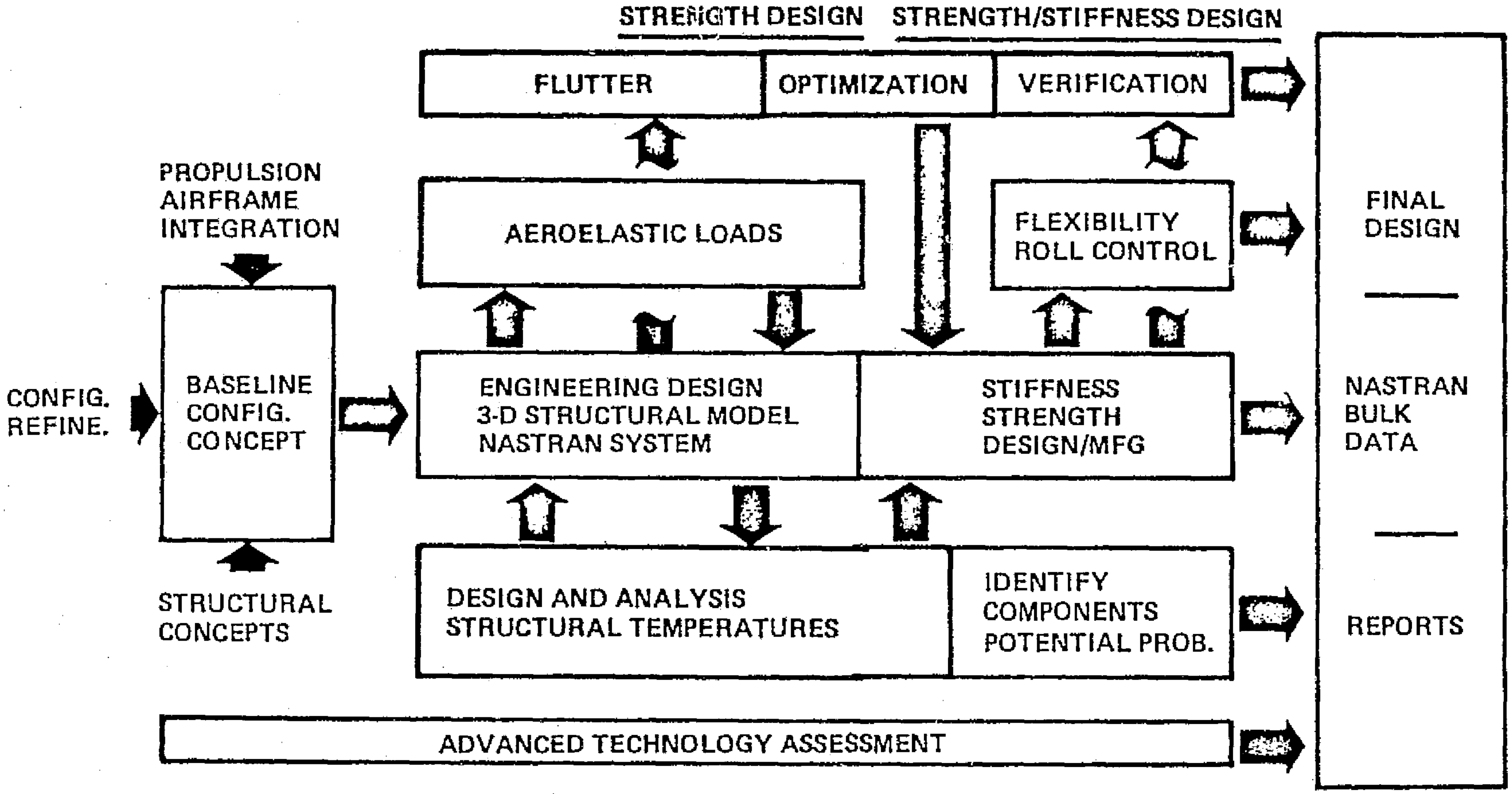

Figure B. Engineering Design and Analyses

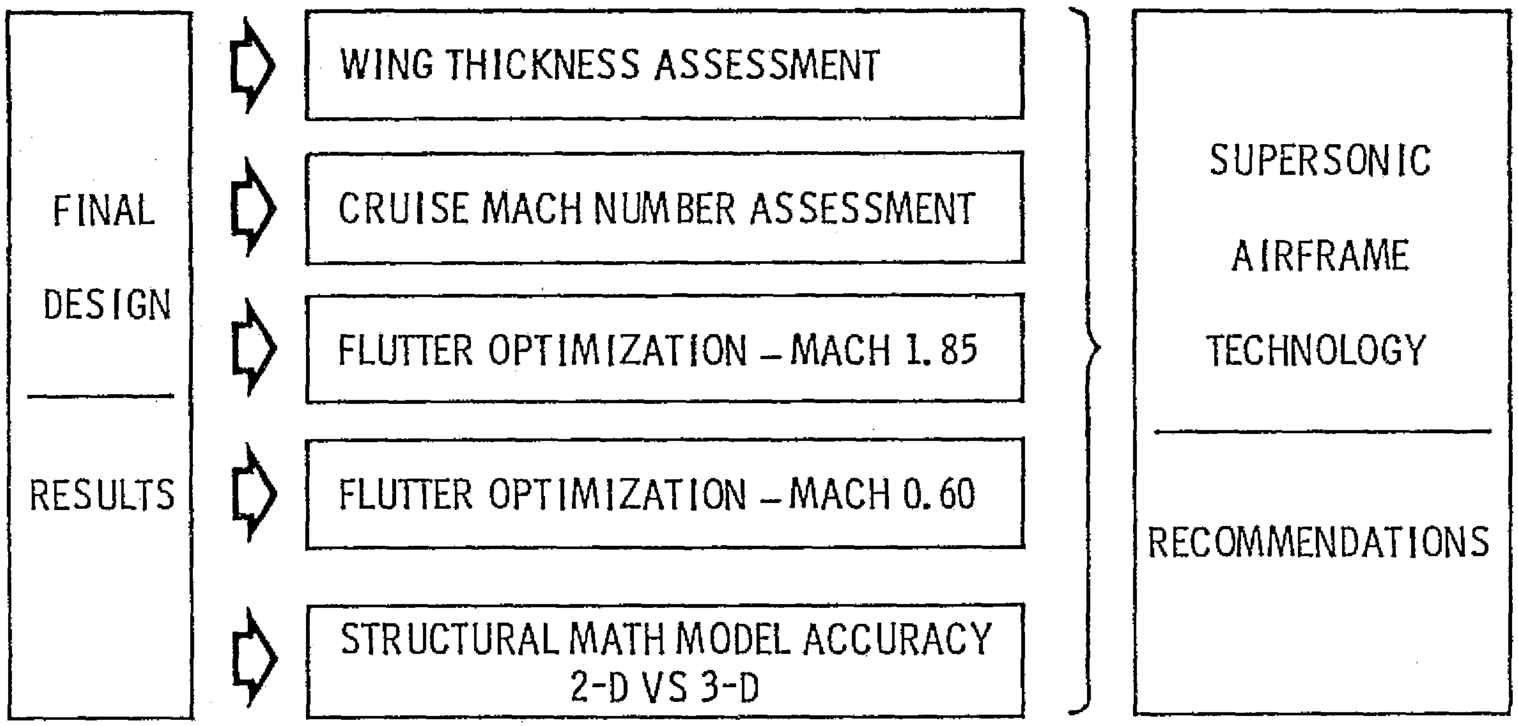

Figure C. Mass Sensitivity Studies

ACKNOWLEDGEMENT

This investigation was conducted under NASA Contract No. NAS1-12288, Study of Structural Design Concepts for an Arrow-Wing Supersonic Transport Configuration. The study was performed within the Science and Engineering Branch of the Lockheed-California Company, Burbank, California, with the participation of the Lockheed-Georgia Company in a composite design subcontract effort. I. F. Sakata was the Project Engineer, and G.W. Davis was the Lead Engineer. The other contributors to the program are acknowledged at each section.

Mr. J. C. Robinson of the Design Concepts Section, Thermal Structures Branch, and Dr. E. C. Yates, Jr., Computer Aided Methods Branch, Structures and Dynamics Division, NASA Langley Research Center, Hampton, Virginia, were the Technical Representative of the Contracting Officer (TRCO), and Alternate TRCO, respectively for the project.

SECTION 12

STRUCTURAL CONCEPTS ANALYSIS

by

G.W. Davis, L.I. Guidry, A.C. Jackson and C.C. Richie

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LIST OF SYMBOLS

A	Cross-sectional area
A,B	X and y distances related to model panels
a,b	Plate dimensions in x and y directions
BL	Buttock line
b	Width
C	Circumferential distance; extreme fiber distance
D	Shell diameter
E	Young's modulus
F	Allowable stress
FS	Fuselage station
f	Stress
feq.	Equivalent stress
f_x, f_y, f_{xy}	Inplane stresses associated with the x-y plane
h	Height
I	Area moment of inertia
K_Q, K_T	Fatigue quality indices
L	Length
M	Bending moment
MS	Margin of safety
N_x, N_y, N_{xy}	Normal and shearing forces per unit distance in the middle surface of plate or shell
N_θ, N_ϕ	Membrane forces per unit length of principal normal sections of a shell
n	Ramberg-Osgood parameter
n_z	Vertical inertia load factor

LIST OF SYMBOLS (CONTINUED)

P	Axial load
p	Pressure; stiffener pitch
Q	Static moment of area
q	Pressure
R	Radius
RT	Room temperature
S	Honeycomb core cell size
$T_{av.}$	Average temperature
t	Thickness
t_s	Skin thickness
t_1, t_2	Interior and exterior face sheet thickness of honeycomb-core sandwich panels
$\overline{t}$	Equivalent panel thickness
V	Vertical shear; velocity
W	Weight
ω	Equivalent panel unit weight
ω_c	Weight of the core of the honeycomb-core of the honeycomb-core sandwich
α	Mean coefficient of thermal expansion
ΔT	Temperature difference
$\epsilon_x, \epsilon_y, \epsilon_{xy}$	Inplane strains associated with the x-y plane
Θ	Semi-apex angle of chordwise stiffened panel concepts
ρ_c	Honeycomb-core density; composite material density
ρ_M	Metal material density

SECTION 12

STRUCTURAL CONCEPTS ANALYSIS

INTRODUCTION

The design of an economically viable supersonic cruise aircraft requires the lowest attainable structural-mass fraction commensurate with the selected near-term structural-material technology. To achieve this goal of minimum structural-mass fraction, various combinations of promising wing and fuselage primary structure were analysed for the load-temperature environment applicable to the arrow-wing configuration. This analysis was conducted in accordance with the design criteria specified in Section 4 and included extensive use of computer-aided analytical methods to screen the candidate concepts (Task I) and select the most promising concept(s) for the in-depth structural analysis (Task II).

Structural Design Concepts

Both wing and fuselage primary load-carrying structural concepts were investigated for application to the arrow-wing configuration. For the wing analysis structural arrangements were investigated that included candidate surface panels. spars and ribs, and the associated non-optimum factor. These candidate concepts are characterized by the type of wing primary load-carrying arrangement (i.e., chordwise, spanwise, and monocoque) and are shown in Figure 12-1. Similarly, the fuselage analysis included the investigation of the major weight components associated with fuselage design, i.e., the shell and frame. Figure 12-2 contains a list of the panel and frame concepts evaluated. Although the sandwich shell was recognized to have potential benefits for (1) structural mass reduction, (2) sonic-fatigue resistance, and (3) reduced sound and heat transmission over the panel concepts shown in the figure, it was not included as part of the study. The results (Appendix A) of the structural assessment performed to quantify the potential mass savings of the honeycomb sandwich fuselage for a near-term supersonic cruise aircraft, indicated a weight disadvantage for the sandwich shell because of the parasitic weight of the titanium alloy core and aluminum alloy braze material.

For both the wing and fuselage analysis, candidate metallic and composite material were considered. The metals included representative Alpha-Beta (Ti-6Al-4V) and Beta (Beta C) titanium alloys. For the composite materials, Boron/polyimide, Boron/aluminum, and Graphite/polyimide reinforced structure were evaluated. A more detailed description of these structural-material concepts and their corresponding fabrication methods and design parameters (constraints) are presented in Sections 1, 7, and 8, respectively.

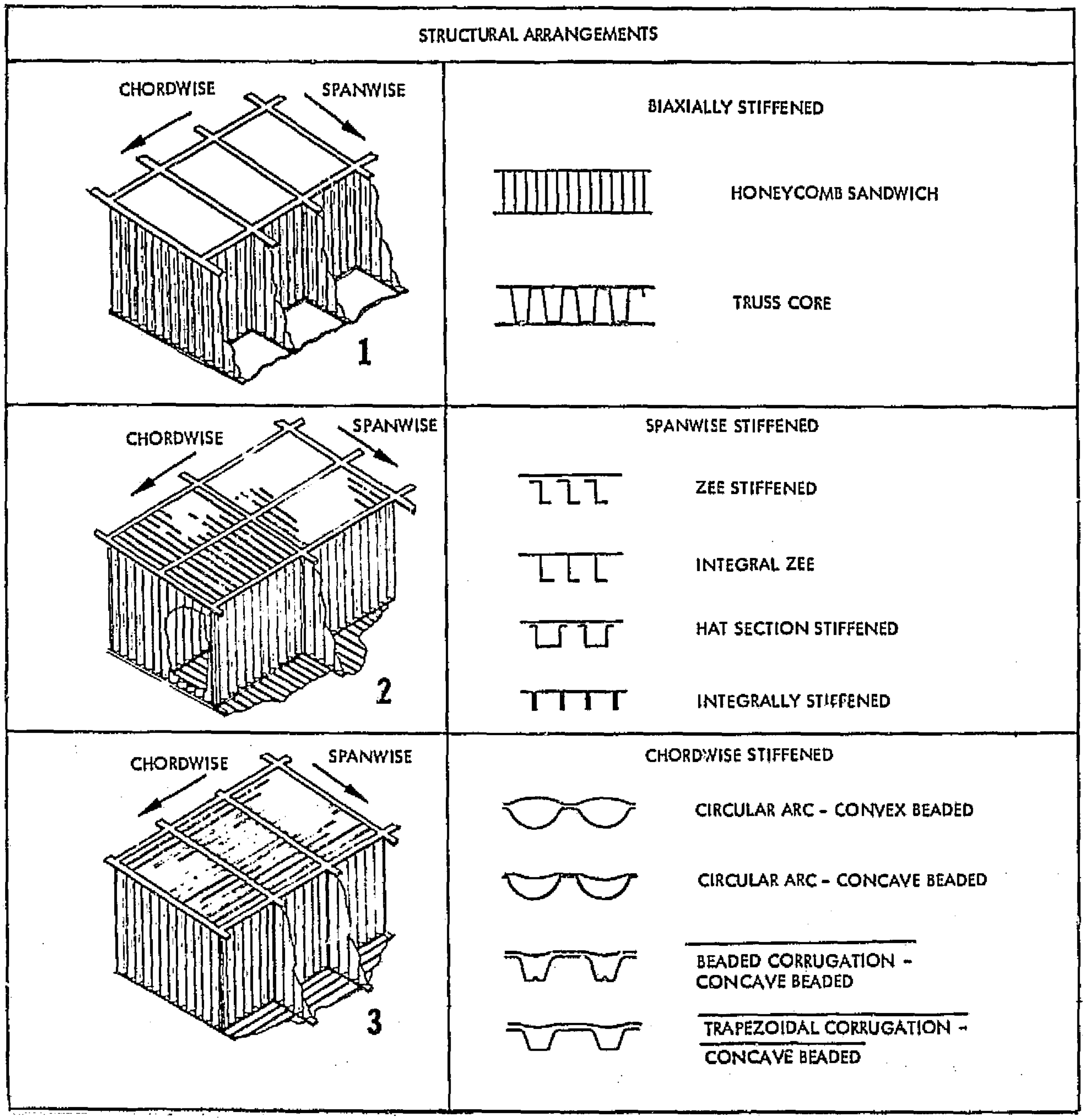


Figure 12-1. Candidate Wing Structural Arrangements

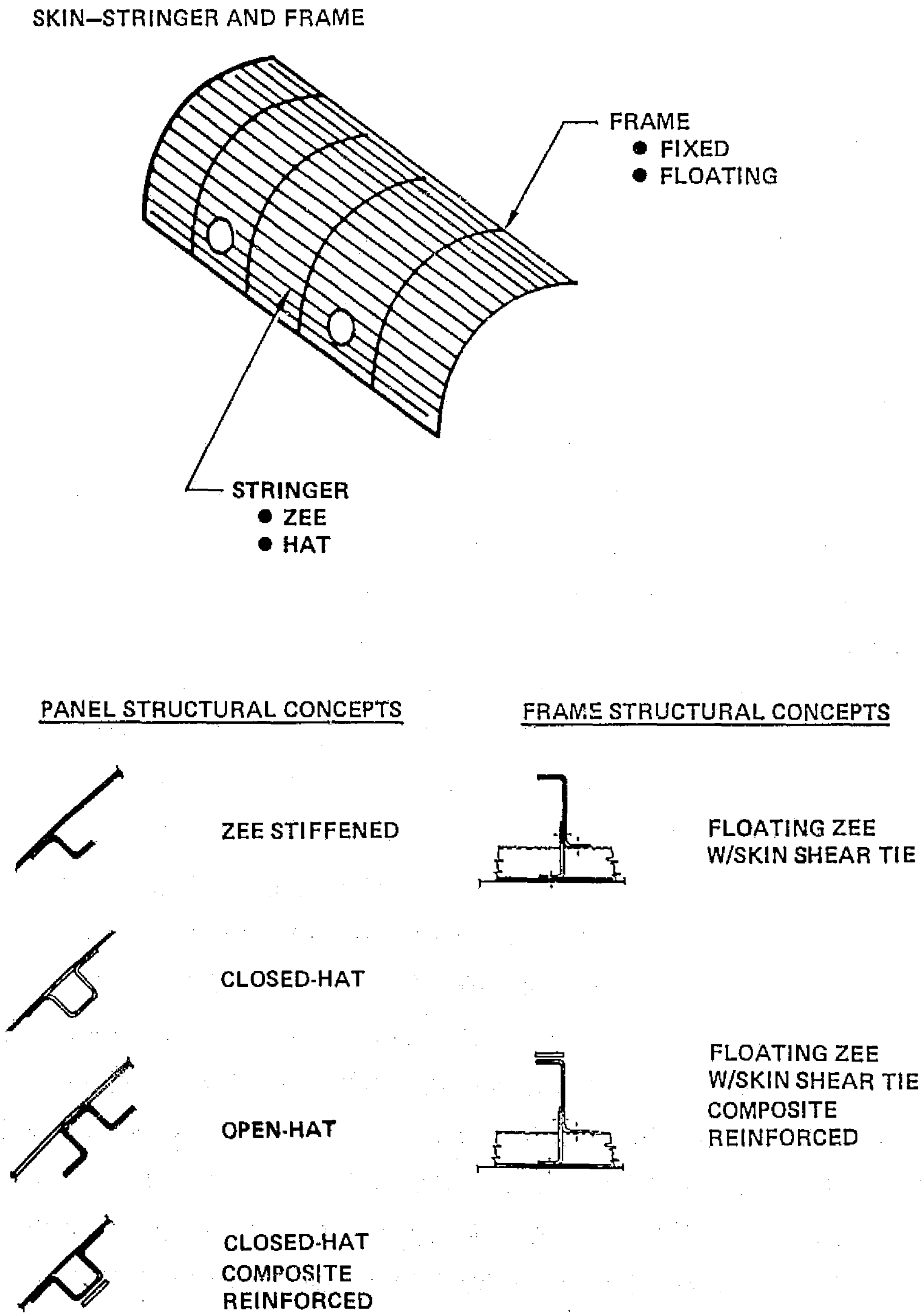


Figure 12-2. Candidate Fuselage Structural Arrangements

Point Design Regions

The basis for the structural-material evaluation was the definition of the candidate structural components and the load-temperature environment at selective wing and fuselage regions. These regions, hereafter referred to as point design regions, are described in the following text.

Wing Point Design Regions. - The location of the wing point design regions are shown in Figure 12-3 and includes the six-regions which are displayed on the wing planform of the structural model. These regions are identified by the NASTRAN panel element numbers used for the finite element model (Section 9). Representative structure is specified at each of these locations and include a definition of the upper and lower surface panels, typical rib and spar structure, and the associated non-optimum factors. These regions were selected as representative of wing critical design regions. A description of these regions is as follows:

Forward wing box - Point design region 40322 is located forward of the main landing gear in a fuel tank region. This area is characterized as basically transmitting pressure loads with low load intensities with respect to wing bending loads.

Aft box region - Point design regions 40236, 40536, and 41036 are located in the wing aft box with 40236 and 40536 located in fuel tanks and 41036 in a dry bay region. In general, these areas represent regions of high spanwise load intensities and variable chordwise load intensities due to wing bending. The chordwise load intensities on region 40236, most inboard regions, reflect the influence of fuselage body bending while the outboard region 41036 displays the effect of the wing tip load redirection.

Wing tip region - Dry bay regions 41316 and 41348 are located approximately at the root and mid-span of the wing tip. These areas are characterized by high load intensities indicative of the aero-elastic effect on this flexible region.

Fuselage Point Design Regions. - Four point design regions were selected as representative of the actual fuselage design. These regions are shown in Figure 12-4 and are located at fuselage stations 750, 2000, 2500, and 3000 for the Task I analysis.

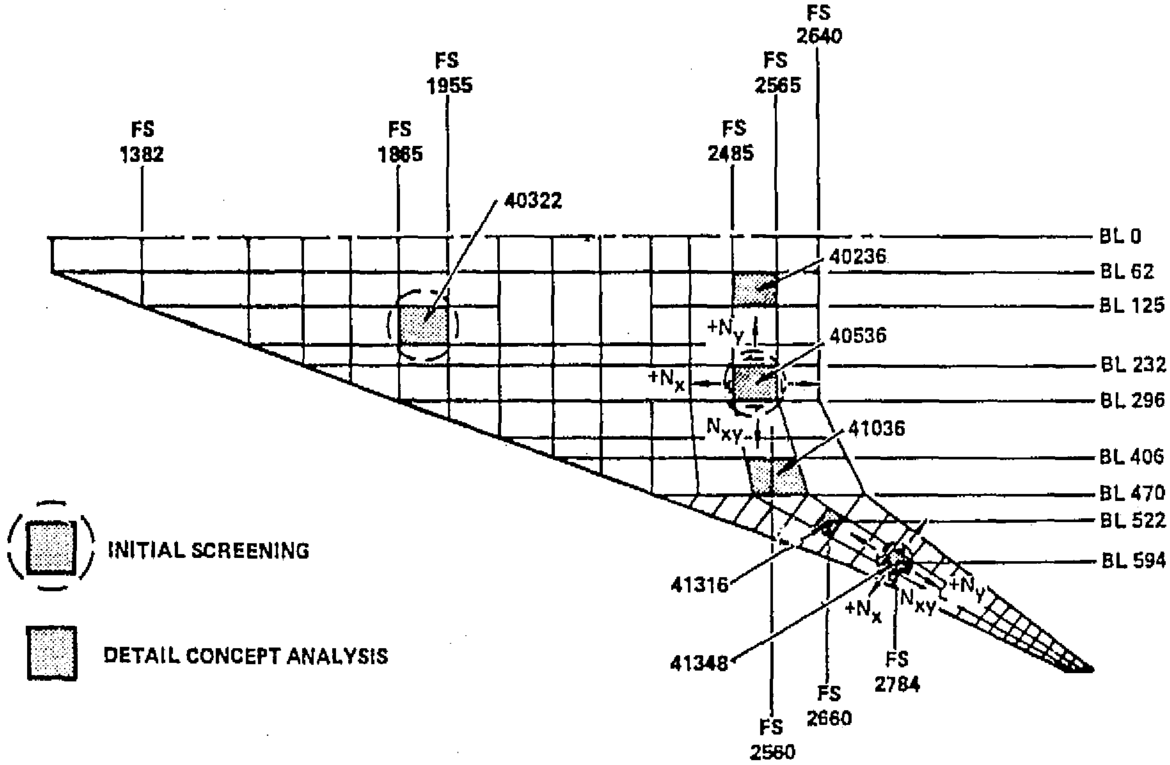


Figure 12-3. Definition of Wing Point Design Regions

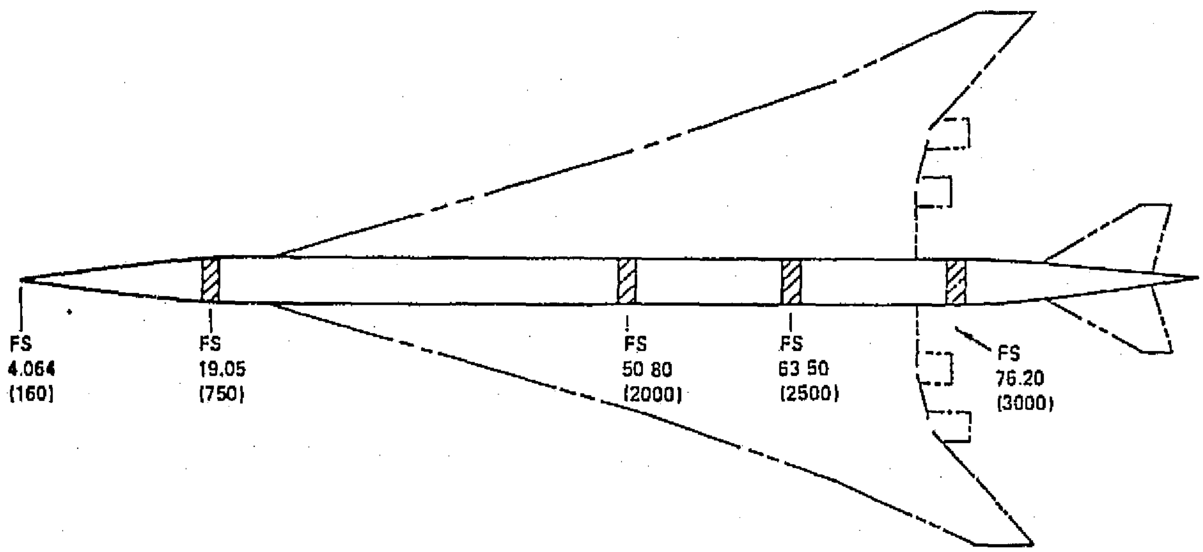


Figure 12-4. Definition of Fuselage Point Design Regions - Task I

For Task II, slight changes in these locations where required to reflect the revisions incorporated on the finite-element model and these changes are presented in the Task II introductory text. Conventional structure composed of skin/stringer panels and sheet metal frames were selected for these regions. The panel concepts were varied to reflect the specific design being evaluated. These regions were selected as typical of the critical design regions on the fuselage and, in general, classified as follows:

Fuselage Forebody (FS 750) - Generally characterized as fatigue-designed structure with low load intensities due to fuselage bending.

Fuselage Centerbody (FS 2000 and 2500) - Wing/fuselage regions subjected to maximum body bending and wing spanwise loads.

Fuselage Aftbody (FS 3000) - High body bending and torsion loads with regions subjected to a high acoustic environment.

Fuselage point design regions located at FS 2000 and FS 2500 are coincidental with the wing forward box and aft box point design regions.

Point Design Environment

The load-temperature environment was defined for each wing and fuselage point design region in support of the specific task being conducted. A detail description of this environment is presented in Section 11, Point Design Environment, and in general included:

- The load intensities and thermal strains from the NASTRAN internal load solution.
- The normal loads acting at each region, considering both aerodynamic pressure and fuel inertia heads.
- The average component temperatures and gradients associated with the specific structural arrangement.

In addition to the detail description of the point design environment contained in Section 11, each of the enclosed analysis sections contain the point design environment for its critical flight condition(s).

Analytical Methods

Structural analyses were performed on each candidate wing and fuselage concept, Figures 12-1 and 12-2, to define the minimum weight designs and corresponding panel proportions. These analyses were conducted with computer programs which used sound analytical methods and incorporated optimization subroutines for determining the minimum weight design. These programs, formulated for either the direct-search or synthesis-method of structural optimization, generally included the following subroutines: (1) definition of the total inplane stress resultants, (2) calculation of the section properties and stiffnesses, (3) a stress analysis, (4) definition of the allowable stresses, and (5) the optimization procedure.

Chordwise Panel Concepts - A computer program which uses the direct search method, was used to determine the minimum weight designs for chordwise concepts. The analytical methods employed in this program, which is entitled STRUDE II, are reported in Section 12 of Reference 1 and analyzes these concepts for the total inplane stress resultants and normal pressure. In addition, the analysis procedure includes the bending moment attributed to eccentric edge loading, initial deflection due to manufacturing, and bowing caused by a temperature gradient through the panel thickness.

For the compression-combined load condition, the theory is based on the wide-column approach of Reference 2 modified to include bending loads with an interaction equation used to include the shear load. The magnification effects of simultaneously applied axial and transverse loadings (beam column analysis) is included for the compression-load condition but conservatively neglected for the tension condition.

For the combined load condition with an applied tension axial load, the maximum equivalent stresses (f_{eq}) were calculated using the principal stress equation:

$$f_{eq} = \frac{f}{2} \pm \sqrt{\left(\frac{f}{2}\right)^2 + f_{xy}{}^2}$$

where (f) and (f_{xy}) represent the axial and shear stresses respectively. The allowable tensile stresses were based on fatigue allowables commensurate with the panel fatigue quality index (K_Q) for a calculated fatigue life of 1.25×10^5 flight-hours. These stresses were determined by the methods described in

Section 13 and are reported, along with the associated quality index, in the following analysis sections.

Spanwise Panel Concepts - The minimum weight panel designs for the spanwise concepts were determined using the same methods as described for the chordwise concepts.

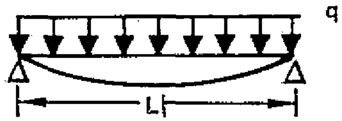
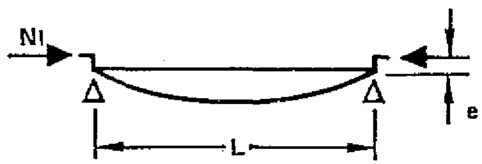
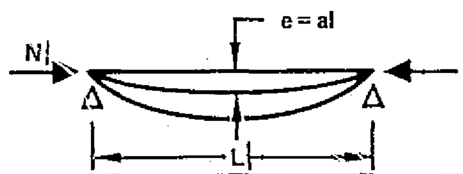
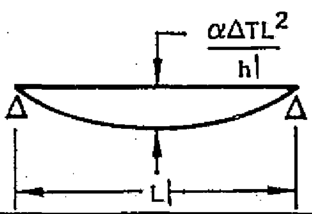
This analysis was conducted using two computer programs, entitled Panel and Fatigue, for the specific compression or tension combined loading condition. As with the chordwise concept, the combination of axial load, shear, and bending moment were included in the analysis. The exception being the method used to account for the bending due to panel edge eccentricity, initial curvature, and thermal bowing where the concept of equivalent design pressure was introduced to include these effects. Table 12-1 presents the equivalent pressure expressions for each type of bending load. Since these values show that the equivalent pressures depend on the panel depth (h) and rib spacing (L) an iterative procedure was included to determine the panel design and equivalent pressure based on a common panel depth.

The same compression and tension design criteria as used for the chordwise concepts were applied to the spanwise concepts. The compression loaded panel designs were based on the local buckling strength of the skin and stiffener elements for combined compressive (due to axial load and bending) and shear loading, whereas, the applied stresses for the tension loaded designs were based on the maximum equivalent stress and compared to the fatigue allowable stress.

Biaxially Stiffened Panel Concepts - The candidate biaxially stiffened panels were analyzed using the STRUDE II computer program, which is described in Section 10 of Reference 1. The two candidate concepts, honeycomb sandwich and truss core, are displayed in Figure 12-1.

The multiple panel loading conditions include biaxial loads, shear, primary bending and secondary or deflection induced bonding. Bending loads include normal pressure, edge eccentricity, initial deflections, coupling eccentricity, and initial curvature due to thermal gradient ($X = \alpha \Delta T / h$). As with the analyses of the preceeding uniaxial stiffened concepts, the beam-column effect was included for the compression combined load condition and conservatively neglected for the tension combined load condition.

TABLE 12-1. EQUIVALENT PRESSURE LOADING – SPANWISE STIFFENED ARRANGEMENTS

LOADING/SUPPORT	MAXIMUM BENDING MOMENT, M	EQUIVALENT PRESSURE, q
• UNIFORM LOADED BEAM SIMPLY SUPPORTED 	$\frac{qL^2}{8}$	q
• EDGE ECCENTRICY 	Ne	$\frac{8Ne}{L^2}$
• INITIAL CURVATURE 	NaL	$\frac{8Na}{L}$
• THERMAL BOWING 	$\frac{N\alpha\Delta TL^2}{8h}$	$\frac{N\alpha\Delta T}{h}$

Analytical Procedures

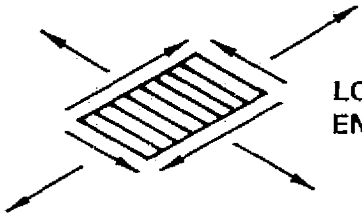
To provide a rational basis for evaluating the weight of the candidate structural arrangements, detail analytical procedures, commensurate with the specific state of design under consideration, were established for conducting the structural analysis. These procedures are described in the following text for each major study task of this program, i.e., Task I, Analytical Design Studies and the Task II Detailed Engineering Studies.

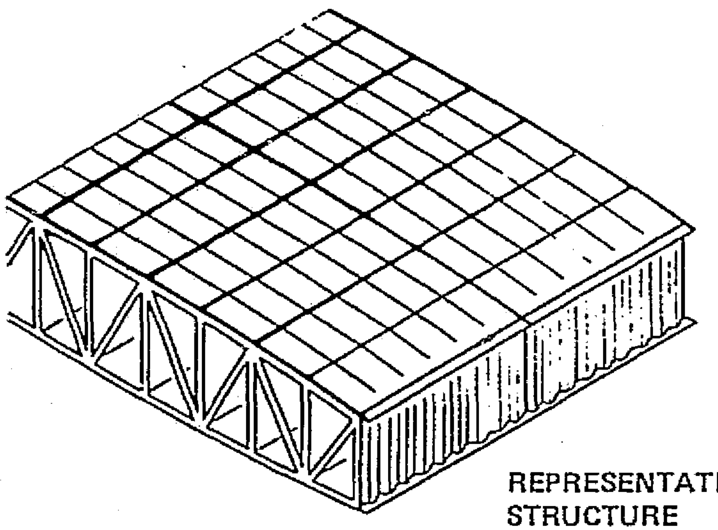
The Task I Analytical Design Studies were conducted in two stages as defined in Figure 12-5 as the Initial Screening and Detail Concept Analysis. The Initial Screening Analysis, in general, consisted of the following steps:

(1) At selected point design regions the load-temperature environments were defined for each general type of wing arrangement and the single fuselage arrangement. For each wing arrangement the basis for the internal loads was the NASTRAN redundant structure analysis solution using 2-D structural models. For the fuselage, existing body shear and bending moment diagrams were used to calculate the theoretical internal load distributions. Aerodynamic heating analysis were conducted to determine the average temperature and gradients on the wing and fuselage primary structure.

(2) A weight/strength analysis was conducted on each of the candidate wing panel and fuselage shell concepts. Panel proportions and unit weights were determined for various rib and spar spacings for the wing surface panel concepts and frame spacings for the fuselage concepts. Computer-aided analytical methods were used to optimize the panels for their most critical tension or compression design condition. For the compressive condition, local and general instability modes were included with plastic deformation taken into account with the use of the Ramberg-Osgood stress-strain relationship. For the tension stress state, the equivalent stress was not allowed to exceed the gross area fatigue allowable commensurate with the fatigue quality of the panel under investigation. In addition to the strength analysis, damage tolerance analyses were conducted at selective locations for each of the candidate panel concepts and the results are reported in Section 13.

POINT DESIGN





DEPTH OF ANALYSIS

INITIAL SCREENING	DETAIL CONCEPT
● ALL STRUCTURAL CONCEPTS	● MOST PROMISING CONCEPTS
● THREE POINT DESIGN REGIONS	● ADDITIONAL POINT DESIGN REGIONS
● ANALYSIS ULTIMATE LOADS FATIGUE DAMAGE TOLERANCE	● ANALYSIS ULTIMATE LOADS FATIGUE SONIC FATIGUE FLUTTER DAMAGE TOLERANCE
● MASS ANALYSIS	● MASS ANALYSIS

Figure 12-5. Task I Analytical Design Procedure

(3) As a result of the panel weight/strength analysis each panel concept within a general arrangement was ranked in accordance with weight, e.g., with reference to the chordwise stiffened wing arrangement shown in Figure 12-1, the circular arc-convex beaded panel weighed less than circular arc-concave beaded, corrugation-concave beaded and the beaded corrugation-concave beaded concepts. From this ranking the most promising panel concept from each wing and fuselage arrangement was selected for further evaluation in the next stage of the Task I analysis, the Detail Concept Analysis.

The Detail Concept Analysis was conducted on typical wing box and fuselage structure. In addition to the surface panels, the investigation included an evaluation of the substructure, i.e., the ribs and spars for the wing box and the frames for the fuselage segment. The analytical approach was shown previously in Figure 12-5 and was conducted in accordance with the following procedure:

(1) Additional wing and fuselage point design regions were selected and their specific point design environment defined. The load-temperature environment was based on the same NASTRAN redundant structure analysis solutions and aerodynamic heating calculations performed for the initial screening analysis.

(2) Each of the wing arrangements and the fuselage arrangement were subjected to a weight/strength analysis which included a further evaluation of the most promising panel concepts surviving the initial screening analysis and typical substructure applicable to each basic arrangement. As discussed in the Initial Screening procedure, the components were analyzed for the most critical ultimate design condition with a fatigue cut-off stress being used for the tensile stress-state condition. Additional analyses were conducted on those arrangements which included basic airplane flutter, damage tolerance, and sonic fatigue. These results are reported in Sections 10, 13, and 14, respectively.

(3) The results of the Detail Concepts Analysis were the weight ranking of the basic wing arrangements and the fuselage arrangement. Weight comparisons were made by reviewing the point design unit weights as well as the total airplane weight, reported in Section 15. In addition, these arrangements were evaluated for cost and performance, reported in Sections 16 and 17.

As a result of these evaluations, a hybrid wing design (combination of structural-material concepts) and a conventional fuselage design were selected as the best airplane structural arrangement warranting further evaluation in the Task II Detail Engineering Studies.

The Task II Detailed Engineering Studies were conducted using the least weight hybrid arrangement resulting from the Task I analysis with its corresponding wing rib and spar spacings and fuselage frame spacings. The major structural components of this arrangement were subjected to an in-depth structural analysis consisting of the following steps:

(1) A 3-D structural model was established using the stiffnesses representative of the strength-sized hybrid arrangement and a NASTRAN redundant structure analysis solution obtained. Using these results the wing and fuselage point design environments were redefined.

(2) The structural concepts at each of the six wing and four fuselage regions were subjected to point design analysis which included evaluation for ultimate loads, load-fatigue, sonic-fatigue, and damage tolerance. In addition, airplane vibration and flutter analyses were conducted.

(3) The definition of airplane stiffnesses resulting from the above structural analysis were compared to those values input in the 3-D structural model described in Step (1). The stiffnesses were generally in good agreement except for the highly elastic wing tip where the required stiffnesses dictated by the aeroelastic and flutter effects were in considerable disagreement with the initial model input values. Because of this difference in wing tip stiffness, the model input data (element properties) described in Step (1) were altered to reflect the new strength and stiffness requirements and a new NASTRAN solution was conducted.

(4) The aeroelastic loads, internal loads, and vibration and flutter analyses were performed using the data generated from the new NASTRAN solution, i.e., structural influence coefficient and stiffness matrices. The mass matrices used in the above analyses were revised to reflect the ammended model stiffness.

(5) In general, good agreement in load intensities and displacement were obtained in the strength-designed regions between the design cycle conducted using the strength-sized model and those of the strength/stiffness model. This convergence precluded the need for any major strength reanalysis.

(6) The unit weights defined at the strength-designed and stiffness-designed regions were used to define a group weight statement and total weight for the baseline Final Design airplane, see Section 15.

The results of the weight/strength analyses described in the above procedures are presented in this section, whereas, the vibration and flutter, damage tolerance (fatigue and fail-safe), sonic-fatigue (acoustics), structural design loads, and mass analyses are reported in their respective sections of this report. Specifically for the weight/strength analysis, these procedures are described for each wing and fuselage arrangement investigated and are presented in the order or occurrence in which they were conducted for this study. Additional introductory remarks and data are presented for the Task II analyses to maintain continuity for the reader.

CHORDWISE STIFFENED WING ARRANGEMENT - TASK I

An Initial Screening and a Detailed Concept Analyses were conducted on the chordwise wing structural arrangement. During the Initial Screening Analysis two candidate structural-materials (both metal) and four candidate panel concepts were investigated. The four panel concepts are presented in Figure 12-6. Also included on this figure is a typical wing box segment depicting the major wing components included in the Detailed Concepts Analysis. Those components considered in this analysis are: the surface panels, spar caps and webs, rib caps and webs, and the appropriate non-optimum factors.

Fabrication limits for the chordwise panels and closures are summarized in Figure 12-7 with a detailed description of this data contained in Section 7, Materials and Producibility Section.

The basis for the structural analysis was the internal loads resulting from the NASTRAN redundant structural analysis solution. A 2-D structural model with flexibilities representative of a typical chordwise stiffened wing was used for this solution. These internal loads in conjunction with the applied pressures (aerodynamic and fuel) and temperatures defined the point design environment for these chordwise panels. Table 12-2 contains the most critical Task I point design environment.

Chordwise Initial Screening

The chordwise initial screening analysis was conducted in two parts, which were: (1) a material tradeoff study to select the most promising material system and (2) a detail analysis to screen the candidate panel concepts and select the least-weight concept for further valuation.

The initial material tradeoff study was conducted using a representative Beta (Beta C) and Alpha-Beta (6Al-4V) titanium alloys. This tradeoff study was conducted by strength-sizing both materials for the trapezoidal corrugation-panel concept (Figure 12-6) using the point design environment specified for region 40536 (Table 12-1). The results of this study indicated the Alpha Beta (6Al-4V) alloy was the least-weight concept and this material system was selected for application to the candidate panel concepts for the screening analysis.

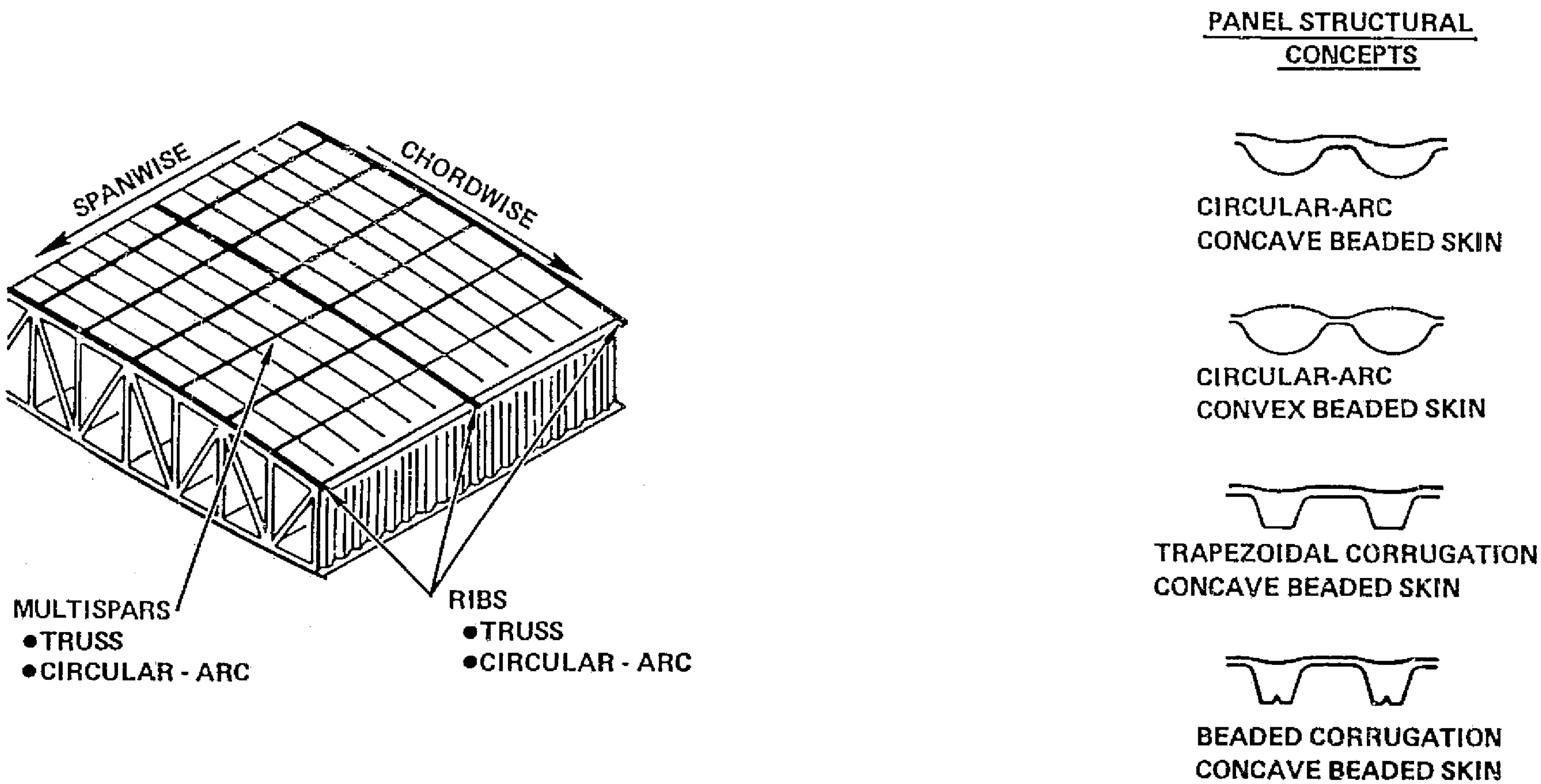


Figure 12-6. Chordwise Stiffened Wing Structural Arrangements

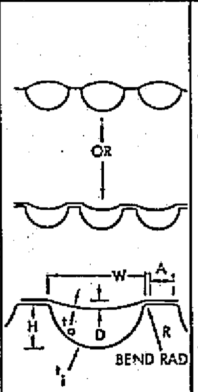
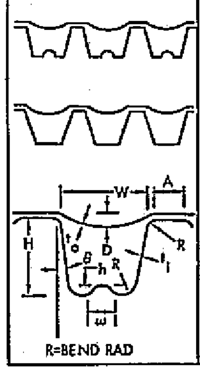
CONCEPT	FACE MATL.	INNER STIFF MATL.	SKIN GAUGES		SHEET SIZE (INS)	OUTER PANEL SIZE (FT)	INNER PANEL SIZE	FORMING LIMITS: OUTER SKIN: METHOD	FORMING LIMITS: OUTER SKIN: $\frac{D}{W}$ (MAX.)	FORMING LIMITS: INNER SKIN: METHOD	INNER SKIN: $\frac{H}{W}$	INNER SKIN: $\frac{h}{w}$	INNER SKIN: MIN. BEND RADI	INNER SKIN: A	FOREIGN OBJECT DAMAGE LIMITS	METHOD OF ATTACH. INNER TO OUTER SURFACE	CONTOUR LIMITS	COMMENT
	TI-6Al-4V AN-NEALED	TI-6Al-4V AN-NEALED	MIN	.010	35 x 144	15 x 35 WELD FROM SMALLER SIZE SHEETS	THREE BEAD WIDTH FORMING LIMITATION. WELD SECTIONS TO GIVE FULL WIDTH PANEL	PREFORM (COLD) PLUS VACUUM HOT FORM FULL WIDTH PANEL 1450°F	.10	HOT FORM THREE BEAD WIDTH FORMING LIMIT TO AVOID EXCESSIVE THINNING. WELD END CLOSURES VACUUM HOT-FORM (1450°F) – FULL WIDTH PANEL.	.48 MAX		3t	MIN. .6 LIMIT SET FOR FRACTURE SPAR CLIP MECH. ATTACH	t_i = .010 t_o = WING UPPER SURFACE .015 WING LOWER SURFACE .020	WELD BOND CAPILLARY ACTION FOR MIN. WT.	NO PRACTICAL LIMIT	SKINS WILL HAVE K_T INDUCED BY WELD QUALITY INDEX K_Q = 4 ASSUMED
			MAX	.125	60 x 200													
	TI-6Al-4V AN-NEALED	BETA ALLOY STA	MIN	.010	24 x 144	15' x 35' WELD FROM SMALLER SIZE SHEETS	AS ABOVE			COLDFORM THREE BEAD WIDTH WELD END CLOSURES VACUUM HOT-FORM (1450°F) – FULL WIDTH PANEL STA (1100°F)	.40 MAX							
			MAX	.125	60 x 200													
	TI-6Al-4V AN-NEALED										$\frac{H}{W}$	$\frac{h}{w}$	θ AND R					
		TI-6Al-4V AN-NEALED	MIN	.010	35 x 144	15' x 35' WELD FROM SMALLER SIZE SHEETS	THREE STIFFENER WIDTH FORMING LIMIT WELD SECTIONS TO GIVE FULL WIDTH PANEL	AS ABOVE	AS ABOVE	AS ABOVE FOR TI 6Al-4V ANNEALED	.8 MAX		3° MIN	AS ABOVE	AS ABOVE			
			MAX	.125	60 x 200													
		BETA ALLOY STA	MIN	.010	24 x 144	15' x 35' WELD FROM SMALLER SIZE SHEETS				AS ABOVE BETA ALLOY STA		.4 MAX	BEND RAR $R=3t_i$					
			MAX	.125	60 x 200													

Figure 12-7. Fabrication Limits – Chordwise Stiffened Surface Panels

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TABLE 12-2. WING POINT DESIGN ENVIRONMENT, CHORDWISE ARRANGEMENT - TASK I

WEIGHT = 690 X 10^3 LB

CONDITION 31 ; MACH NO. = 1.25; n_z = 2.5

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION											
			40236		41036		41316		40536		41348		40322	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	N_x	LB/IN	[illegible]	-258	-1,435	1,435	571	-571	-1355	1355	-1433	1433	455	-455
	N_y	LB/IN	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-14,375	14,375	-10,800	10,800	-1,063	1,063
	N_{xy}	LB/IN	[illegible]	[illegible]	2,237	[illegible]	4,807	4,807	[illegible]	[illegible]	2,483	2,483	120	120
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	[illegible]	0	0	0	0	0
PRESSURE	AERO	PSI	-3.33	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-6.07	[illegible]	-1.47	[illegible]
	FUEL	PSI	[illegible]	[illegible]	0	0	0	0	[illegible]	[illegible]	0	0	-6.86	-9.00
	NET	PSI	[illegible]	-10.14	-1.27	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-8.33	[illegible]
TEMPERATURE	T_{AV}	°F	143	134	[illegible]	204	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	164	150
	ΔT	°F	[illegible]	[illegible]	-45	[illegible]	[illegible]	-51	[illegible]	[illegible]	[illegible]	[illegible]	-123	[illegible]

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

WEIGHT = 660 X 10^3 LB

CONDITION 20 (START OF CRUISE); MACH NO. = 2.7; n_z = 2.5

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION											
			40236		41036		41316		40536		41348		40322	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	N_x	LB/IN	-22	22	-541	541	[illegible]	-681	[illegible]	[illegible]	-137	137	-122	122
	N_y	LB/IN	[illegible]	[illegible]	-4,921	4,921	-10,062	10,062	-3,707	3,707	-5,564	5,564	-866	866
	N_{xy}	LB/IN	-601	601	-1,235	1,235	2,202	-2,202	-1,187	1,187	664	-664	-97	97
THERMAL STRAIN	ϵ_x	IN/IN	-282×10^{-6}	282×10^{-6}	94×10^{-6}	-94×10^{-6}	13×10^{-6}	-13×10^{-6}	-40×10^{-6}	40×10^{-6}	0	0	-632×10^{-6}	632×10^{-6}
	ϵ_y	IN/IN	-201×10^{-6}	201×10^{-6}	15×10^{-6}	-15×10^{-6}	19×10^{-6}	-19×10^{-6}	-91×10^{-6}	91×10^{-6}	1×10^{-6}	-1×10^{-6}	-939×10^{-6}	939×10^{-6}
	ϵ_{xy}	IN/IN	426×10^{-6}	-426×10^{-6}	-21×10^{-6}	21×10^{-6}	116×10^{-6}	-116×10^{-6}	-280×10^{-6}	280×10^{-6}	1.25×10^{-6}	-1.25×10^{-6}	-17.4×10^{-6}	17.4×10^{-6}
PRESSURE	AERO	PSI	-1.70	-0.74	-1.25	0.38	-1.65	1.00	-1.47	-0.36	-1.29	1.04	-1.67	-0.17
	FUEL	PSI	-6.42	-7.84	0	0	0	0	-6.00	-7.11	0	0	-6.00	-9.30
	NET	PSI	-8.12	-8.58	-1.25	0.33	-1.65	1.00	-7.47	-7.47	-1.29	1.04	-7.67	-9.47
TEMPERATURE	T_{AV}	°F	204	176	337	339	335	336	212	181	334	347	239	205
	ΔT	°F	-195	-199	-39	-38	-39	-39	-139	-206	-71	-47	-198	-259

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

The chordwise screening analysis was conducted on the four candidate panel concepts which were previously shown in Figure 12-6 and included the following concepts:

- Circular-arc convex beaded skin
- Circular-arc concave beaded skin
- Trapezoidal corrugation-concave beaded skin
- Beaded corrugation-concave beaded skin

These panel concepts were subjected to a weight-strength analysis at three point design regions as shown in Figure 12-3 using the point design environment presented in Table 12-2. This analysis was conducted on all chordwise-stiffened panel concepts, both upper and lower surface panels, for variable spar spacings of 20, 30 and 40-inches with a constant rib spacing of 60 inches.

In summary, the initial screening results indicated the circular-arc convex beaded panel concept afforded the minimum weight design at each of the point design regions. A weight of 1.75 lb/sq.ft. was recorded for the 20-inch spar spacing design at region 40322 and approximately 2.00 lb/sq.ft. for the same designs at regions 40536 and 41348. No consistent ranking, with respect to weight, was noted for the other three panel concepts at the regions investigated.

Material Tradeoff Study - In support of the material selection for the baseline metallic airplane a weight-strength analysis was conducted on representative metals from each general class of allows considered. Ti 6Al-4V(Ann.) and Beta C were chosen as representative of the alpha-beta and beta alloys, respectively. The basic mechanical and physical properties, and fabrication technique of the candidate alloys are presented in the Materials and Producibility Section, Section 7. In addition, the basic design parameters were as defined in Figure 12-7.

The tradeoff study was conducted at point design region 40536 and consisted of sizing both upper and lower surface panel for spar spacings of 20-, 30-, and 40-inches with rib spacing held constant at 60-inches.

The chordwise panel concept investigated for the application of the two materials was the trapezoidal corrugation-concave beaded skin concept. The results of panel sizing for the Beta C material are presented in Table 12-3 and the corresponding Ti 6Al-4V panel data are shown in Table 12-4. Using these results, the upper and

TABLE 12-3. GEOMETRY AND WEIGHT FOR THE TRAPEZOIDAL CORRUGATION PANEL CONCEPT, BETA C MATERIAL – TASK I MATERIAL TRADEOFF STUDY

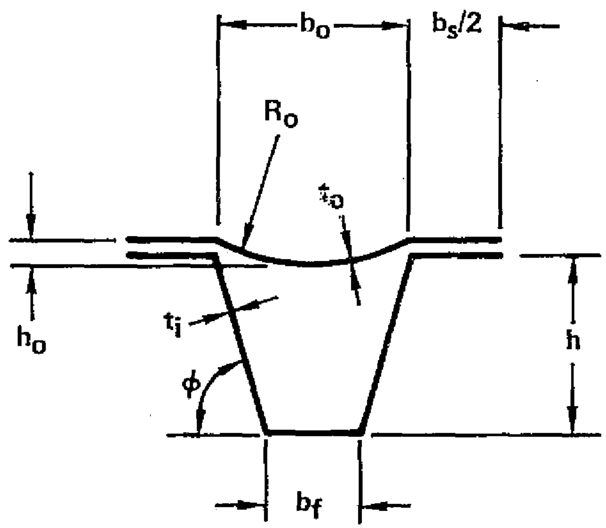
POINT DESIGN REGION	40536					
SURFACE	UPPER			LOWER		
SPAR (m) SPACING (in)	.51 20	.76 30	1.02 40	.51 20	.76 30	1.02 40
DIMENSIONS:						
t_o (in)	.029	.032	.043	.033	.030	.037
t_i (in)	.025	.029	.033	.020	.024	.025
R_o/t_o	53.79	56.88	43.35	59.09	56.33	49.19
ϕ (DEG)	77.47	79.70	79.11	66.37	81.47	90.00
b_o (in)	1.20	1.40	1.50	1.50	1.30	1.40
b_f (in)	0.80	1.00	1.00	0.80	1.00	1.40
h (in)	0.90	1.10	1.30	0.80	1.00	1.20
b_s (in)	0.75	0.75	0.75	0.75	0.75	0.75
MASS DATA:						
$\bar{t}$ (in)	.0730	.0863	.1083	.0629	.0747	.0905
w (lb/ft^2)	1.787	2.113	2.650	1.539	1.828	2.217
CRITICAL CONDITION	31	31	31	31	31	31

PANEL CONCEPT:

TRAPEZOIDAL CORRUGATION–
CONCAVE BEADED SKIN

MATERIAL:

TITANIUM ALLOY BETA C (STA)



$h_o/b_o = 0.10$

TABLE 12-4. GEOMETRY AND WEIGHT FOR THE TRAPEZOIDAL CORRUGATION PANEL CONCEPT, 6Al-4V (ANN.) MATERIAL - TASK I MATERIAL TRADEOFF STUDY

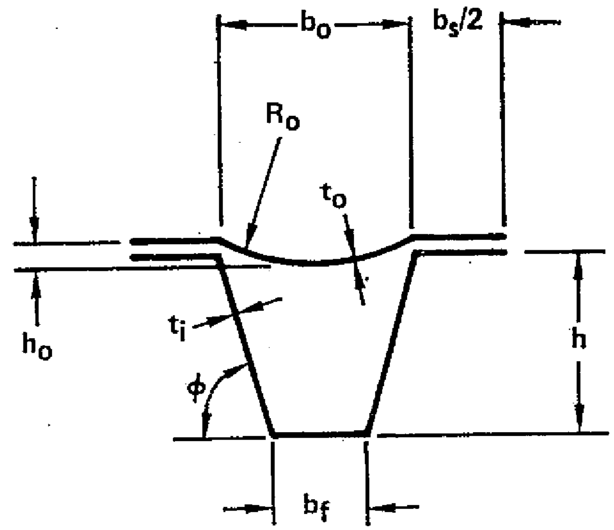
POINT DESIGN REGION		40536					
SURFACE		UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40
DIMENSIONS:							
t_o	(in)	.029	.034	.044	.034	.030	.038
t_i	(in)	.026	.028	.030	.020	.024	.024
R_o/t_o	—	58.28	57.35	44.32	57.35	56.33	51.32
ϕ	(deg)	72.64	77.20	83.42	63.43	81.47	87.80
b_o	(in)	1.30	1.50	1.50	1.50	1.30	1.50
b_f	(in)	0.80	1.00	1.20	0.70	1.00	1.40
h	(in)	0.75	1.10	1.30	0.80	1.00	1.30
MASS DATA:							
$\bar{t}$	(in)	.070	.084	.106	.063	.075	.089
w	($1b/ft^2$)	1.622	1.946	2.435	1.461	1.720	2.059
CRITICAL CONDITION		31	31	31	31	31	31

PANEL CONCEPT:

TRAPEZOIDAL CORRUGATION–CONCAVE BEADED SKIN

MATERIAL:

TITANIUM ALLOY 6Al-4V (ANN.)



$h_o/b_o = 0.10$

$b/2 = .375$

$h_o/b_o = 0.10$

lower surface panel weights were plotted for comparison purposes and are shown in Figure 12-8. With reference to this figure, the Beta C design panel is approximately seven-percent heavier for the lower surface panel and ten-percent heavier on the upper surface. The largest weight variation occured on the compression design upper surface panel and can best be explained by a comparison of the compressive buckling weight index, $\rho/\eta\ (E_c)^{\frac{1}{2}}$. The terms in this expression are: ρ is the material density, η is the plasticity correction factor, and E_c is the compression modulus. For stresses within the elastic region ($\eta = 1$), the compressive buckling weight index for Beta C is approximately twelve-percent higher than Ti 6Al-4V(Ann.) material in the elastic region.

Panel Screening - The four candidate chordwise wing panel concepts shown in Figure 12-6 were analyzed using the most promising metallic material resulting from the material tradeoff study, 6Al-4V(Ann.) titanium alloy. This analysis was conducted at the three wing point design regions indicated in Figure 12-3 using the corresponding load/temperature environment defined on Table 12-2. The analytical methods used to strength-size the panel concepts were as previously described in the introductory text.

Convex Beaded Panel Concept - The results of the strength analysis conducted on the convex beaded panel concept are presented in Table 12-5. This table displays the panel cross-sectional dimensions, mass data, and the critical design conditions. For the three point design regions the skin thicknesses ranged from 0.015-inches to 0.041-inches, with the minimum design thickness constraints (foreign object damage) active for the 20-inch spar spacing design at region 40322. The wing panel unit weights ranged from 0.80 lb/sq.ft. to 1.60 lb/sq.ft. for the lightly loaded region 40322 and approximately 1.30 lb/sq.ft. to 2.20 lb/sq.ft. for regions 40536 and 41348.

To minimize aerodynamic drag the bead height-to-chord ratio (h/c) was held constant at 0.10 with the flat between beads (b) maintained at 0.75 inches to allow for substructure attachment. In addition, the maximum value of the inner bead semi-apex angle (θ = 87 degrees), commensurate with manufacturing limits of this design, was used in the evaluation of all designs.

Concave Beaded Panel Concept - The panel results for this concept are presented in Table 12-6. For this analysis, the cross-section geometry was subjected to the

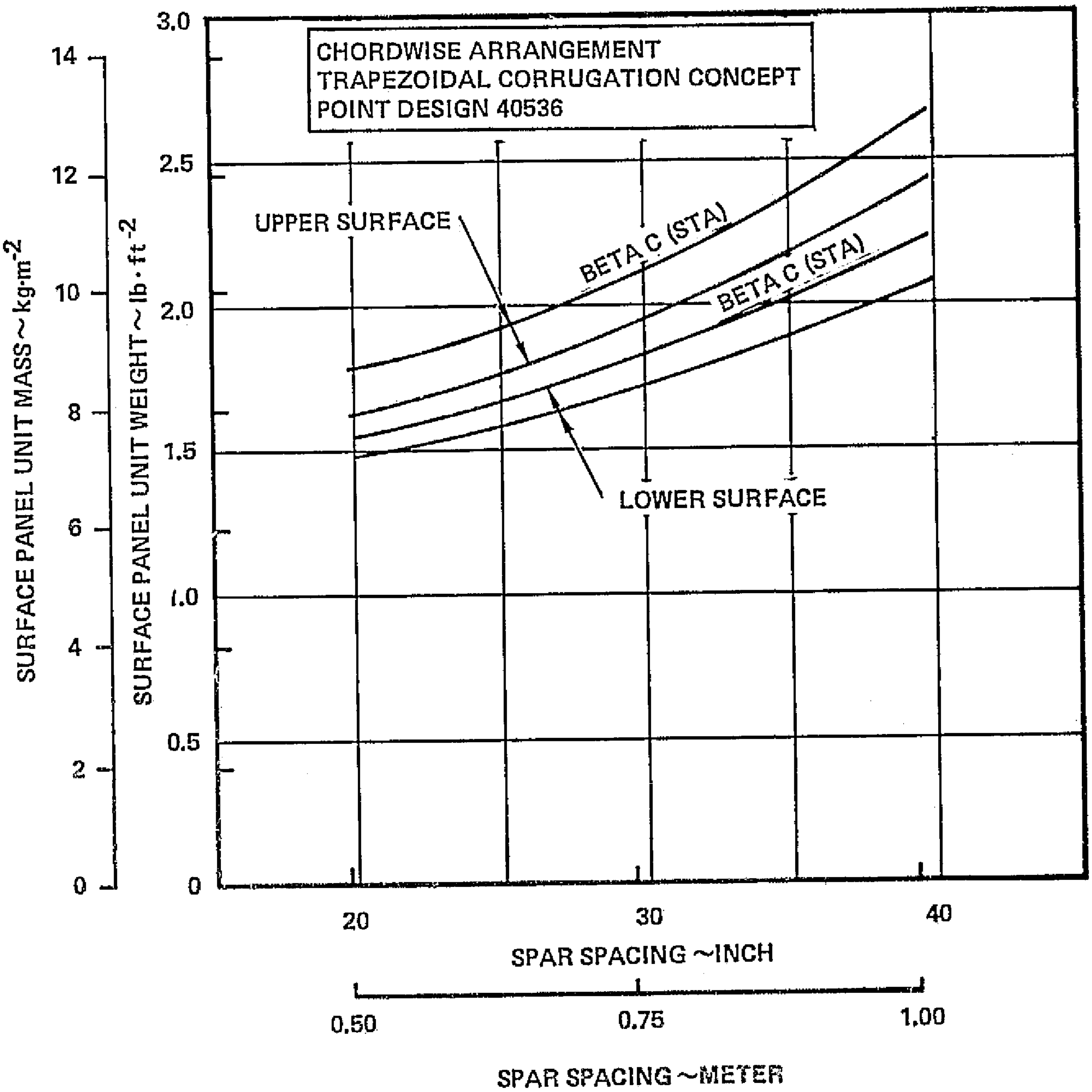
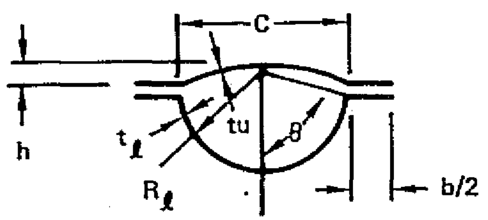


Figure 12-8. Panel Weight of Candidate Metallic Materials, Task I Material Tradeoff Study

TABLE 12-5. PANEL GEOMETRY AND WEIGHT OF THE CIRCULAR ARC-CONVEX BEADED CONCEPT - TASK I CHORDWISE WING ARRANGEMENT INITIAL SCREENING

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_ℓ	(in)	.015	.021	.026	.015	.020	.025	.025	.035	.040	.024	.028	.033	.025	.033	.038	.023	.020	.023
t_u	(in)	.015	.025	.031	.020	.020	.025	.035	.036	.040	.025	.029	.037	.036	.037	.041	.028	.030	.038
R_ℓ	(in)	0.9	1.2	1.4	0.9	1.4	1.8	0.9	1.1	1.4	0.8	1.0	1.4	0.9	1.1	1.4	0.7	0.8	0.9
θ	(deg)	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87
b	(in)	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75
MASS DATA:																			
$\bar{t}$	(in)	.036	.055	.070	.041	.049	.061	.070	.085	.097	.058	.068	.084	.071	.084	.095	.059	.058	.070
w	(lb/ft^2)	.0825	1.263	1.619	0.942	1.120	1.413	1.609	1.965	2.241	1.335	1.570	1.943	1.632	1.925	2.199	1.366	1.328	1.616
CRITICAL CONDITION		20	20	20	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



PANEL CONCEPT:

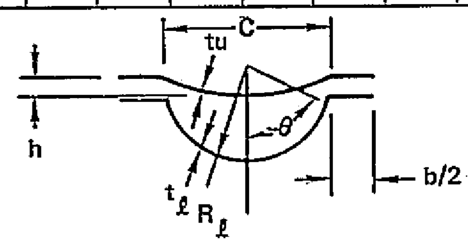
CIRCULAR ARC–CONVEX BEADED SKIN (h/c= 0.10)

TABLE 12-6. PANEL GEOMETRY AND WEIGHT OF CIRCULAR ARC-CONCAVE BEADED SKIN CONCEPT – TASK I CHORDWISE WING ARRANGEMENT INITIAL SCREENING

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_ℓ	(in)	.020	.025	.034	.020	.025	.030	.029	.038	.041	.025	.030	.036	.028	.035	.041	.022	.023	.024
t_u	(in)	.015	.015	.017	.020	.020	.020	.033	.037	.054	.025	.031	.035	.035	.041	.055	.029	.029	.034
R_ℓ	(in)	0.9	1.4	1.7	0.9	1.3	1.7	1.0	1.3	1.7	0.8	1.1	1.4	1.0	1.4	1.6	0.7	0.9	1.1
θ	(deg)	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87
b	(in)	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75
MASS DATA:																			
$\bar{t}$	(in)	.043	.051	.066	.048	.056	.063	.074	.091	.114	.059	.073	.086	.074	.091	.114	.059	.061	.068
w	(1b/ft^2)	0.982	1.165	1.517	1.100	1.279	1.457	1.696	2.099	2.620	1.366	1.688	1.993	1.712	2.101	2.638	1.358	1.405	1.566
CRITICAL CONDITION		20	20	20	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31

PANEL CONCEPT:

CIRCULAR ARC–CONCAVE BEADED SKIN (h/c = 0.10)



same constraints as previously disclosed for the convex beaded concept except for the bead height-to-chord ratio which was held constant at -0.10.

With reference to Table 12-6, the skin thicknesses ranged from 0.015-inches on the upper surface exposed skin at region 40322 to 0.055-inches for the wing tip upper surface skin. The unit weight (lb/sq.ft.) for the panels at region 40322 ranged from 1.00 lb/sq.ft. to 1.50 lb/sq.ft. The unit weights for regions 40536 and 41348 ranged from 1.40 lb/sq.ft. to 2.6 lb/sq.ft.

Trapezoidal Corrugation-Concave Beaded Skin Concept - The results of the strength analysis are summarized in Table 12-7. For this analysis the bead height-to-chord ratio (h/c) of the exposed skin was held constant at -0.10. While the minimum gage thickness and flat distance between beads were identical to those previously discussed for the convex beaded concepts. A sketch of the panel cross-sectional dimensions is contained in the footnotes of the referenced table.

With reference to Table 12-7, skin gages for the beaded skin ranged from 0.020-inches to 0.044-inches while the corrugation thickness varied from 0.019-inches to 0.030-inches. The unit weight of the surface panels at region 40322 ranged from 1.2 lb/sq.ft. to 1.9 lb/sq.ft. The corresponding unit weights for the surface panels at regions 40536 and 41348 varied from 1.3 lb/sq.ft. to 2.4 lb/sq.ft.

Trapezoidal Beaded Corrugation - Concave Beaded Skin - This concept had the same geometric constraints as the previous trapezoidal corrugation concept plus the additional constraints imposed on the corrugation bead. A sketch showing the dimensions of this concept is included in Table 12-8 and indicates the values of the aforementioned geometric constraints.

For the three point design regions the gage thicknesses varied from 0.018-inches to 0.037-inches, corrugation height (h) ranged from 0.70-inches to 1.4-inches, and the bead width (b_o) from 1.1-inches to 1.4-inches. In addition, the corrugation exterior angle (ϕ) varied for 63-degrees to 90-degrees.

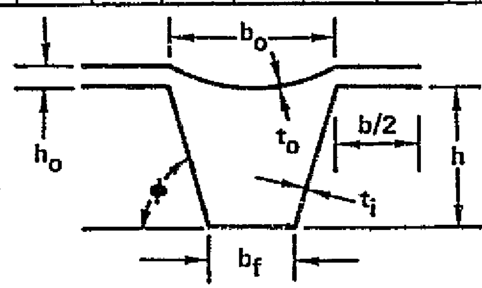
Unit panel weights ranged from 1.2 lb/sq.ft. to 2.2 lb/sq.ft. for region 40322 with regions 40536 and 41348 having slightly higher values, 1.4 lb/sq.ft. to 2.4 lb/sq.ft.

TABLE 12-7. PANEL GEOMETRY AND WEIGHT OF THE TRAPEZOIDAL CORRUGATION-CONCAVE BEADED SKIN CONCEPT - TASK I CHORDWISE WING ARRANGEMENT INITIAL SCREENING

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_o	(in)	.020	.023	.028	.020	.023	.025	.029	.034	.044	.034	.030	.038	.028	.034	.035	.034	.034	.034
t_i	(in)	.019	.023	.028	.019	.022	.024	.026	.028	.030	.020	.024	.024	.024	.029	.034	.018	.020	.024
R_o/t_o	—	97.50	84.78	69.64	84.50	84.78	78.00	58.28	57.35	44.32	57.35	56.33	51.32	46.23	57.35	55.71	57.35	57.35	57.35
ϕ	(deg)	65.77	73.74	74.93	66.04	74.74	75.96	72.64	77.20	83.42	63.43	81.47	87.80	77.47	74.74	79.11	53.13	66.37	65.77
b_o	(in)	1.50	1.50	1.50	1.30	1.50	1.50	1.30	1.50	1.50	1.50	1.30	1.50	1.00	1.50	1.50	1.50	1.50	1.50
b_f	(in)	0.60	0.80	0.80	0.50	0.90	0.80	0.80	1.00	1.20	0.70	1.00	1.40	0.60	0.90	1.00	0.60	0.80	0.60
h	(in)	1.00	1.20	1.30	0.90	1.10	1.40	0.75	1.10	1.30	0.80	1.00	1.30	0.90	1.10	1.30	0.60	0.80	1.00
MASS DATA:																			
$\bar{t}$	(in)	.050	.065	.081	.050	.062	.073	.070	.084	.106	.063	.075	.089	.072	.085	.102	.057	.064	.072
w	(lb/ft^2)	1.158	1.493	1.873	1.156	1.425	1.676	1.622	1.946	2.435	1.461	1.720	2.059	1.664	1.964	2.352	1.322	1.472	1.668
CRITICAL CONDITION		20	20	20	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31

PANEL CONCEPT:

TRAPEZOIDAL CORRUGATION—CONCAVE BEADED SKIN



$h_o/b_o = 0.10$

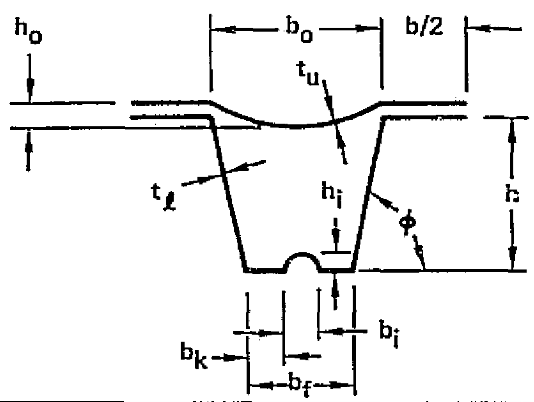
$b/2 = .375$

TABLE 12-8. PANEL GEOMETRY AND WEIGHT OF THE TRAPEZOIDAL BEADED CORRUGATION-CONCAVE BEADED SKIN CONCEP
TASK I CHORDWISE WING ARRANGEMENT INITIAL SCREENING

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_o	(in)	.020	.025	.030	.023	.025	.029	.027	.032	.034	.032	.033	.031	.028	.034	.034	.034	.036	.037
t_i	(in)	.019	.024	.026	.023	.025	.030	.024	.027	.031	.018	.021	.026	.024	.026	.031	.018	.019	.020
R_o/t_o	—	91.00	72.80	60.67	79.13	72.80	58.30	52.96	56.87	53.53	57.88	55.15	58.71	51.07	49.71	53.53	53.53	50.56	49.19
R_i/t_i	—	4.40	4.18	6.43	6.08	7.35	5.01	4.87	8.05	7.01	8.36	10.35	7.71	4.88	8.36	7.01	6.50	6.16	9.19
ϕ	(deg)	65.77	70.02	81.25	71.56	82.87	81.87	77.47	87.40	87.80	72.65	87.14	85.60	77.47	90.00	87.80	63.43	66.37	81.47
b_o	(in)	1.40	1.40	1.40	1.40	1.40	1.30	1.10	1.40	1.40	1.40	1.40	1.40	1.10	1.30	1.40	1.40	1.40	1.40
b_f	(in)	0.50	0.60	1.00	0.80	1.10	0.90	0.70	1.30	1.30	0.90	1.30	1.20	0.70	1.30	1.30	0.70	0.70	1.10
h	(in)	1.00	1.10	1.30	0.90	1.20	1.40	0.90	1.10	1.30	0.80	1.00	1.30	0.90	1.10	1.30	0.70	0.80	1.00
MASS DATA:																			
$\bar{t}$	(in)	.052	.068	.086	.060	.077	.097	.072	.089	.105	.062	.075	.089	.073	.091	.105	.061	.066	.076
w	(1b/ft^2)	1.188	1.563	1.973	1.385	1.782	2.243	1.654	2.047	2.419	1.422	1.737	2.056	1.677	2.106	2.419	1.402	1.520	1.739
CRITICAL CONDITION		20	20	20	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31

PANEL CONCEPT:

TRAPEZOIDAL BEADED CORRUGATION–CONCAVE BEADED SKIN



b_k/b_f = .33
h_i/b_i = .48
h_o/b_o = .10
b/2 = .375

Chordwise Surface Panel Results - The results of the initial screening analyses of the chordwise surface panels are presented in graphic form in Figure 12-9. This figure compares the total panel weight, sum of the upper and lower surface panels weights, at the three point design regions as a function of the variable spar spacing.

With reference to the lightly loaded region 40322, the circular-arc convex beaded concept was the least-weight concept with the trapezoidal beaded corrugation concept the heaviest, e.g., for the 20-inch spar spacing designs the respective values of 1.76 and 2.57 lbs/sq.ft. are noted. The circular-arc concave and trapezoidal corrugation concepts have intermediate values. An exception to this ranking occurs at the 40- spacing were the circular-arc concave beaded concept exhibits a slightly lower value, approximately 2-percent lower, than the convex beaded concept.

For region 40536, the circular-arc convex beaded panel concept was again the least-weight panel concept, less than 3.0 lb/sq.ft. for 20.0-inch spar spacing and slightly over 4.0 lb/sq.ft. at 40-inch spar spacing. These values are indicative of the higher load intensities experienced at this region and hence more efficient designs were obtainable. This trend is indicated in Figure 12-9 by the small weight range exhibited by the four panel concepts.

As with the previous point design regions, Figure 12-9 indicates the circular-arc convex beaded concept is also the least-weight concept for wing tip region 41348. Approximate unit weight values of 3.0 lb/sq.ft. and 3.75 lbs/sq.ft. are noted for spar spacings of 20-inches and 40-inches, respectively. The concept ranking, relative to weight, for this region is (1) circular-arc convex beaded, (2) trapezoidal corrugation-concave beaded, (3) circular-arc concave beaded, and (4) trapezoidal beaded corrugation.

To provide further credence to the selection of the least-weight panel concept a supplemental analysis was conducted to assess any possible changes in panel concept ranking when substructure is incorporated in to the design. This wing box analysis was conducted at the highly-loaded aft box region 40536 and included sizing typical substructure associated with each panel concept. The results of this analysis are contained in Table 12-9 and for comparison purposes displayed graphically in Figure 12-10. This analysis indicates the design using the circular-arc convex beaded panel results in the least-weight wing box. Approximate unit box weights of

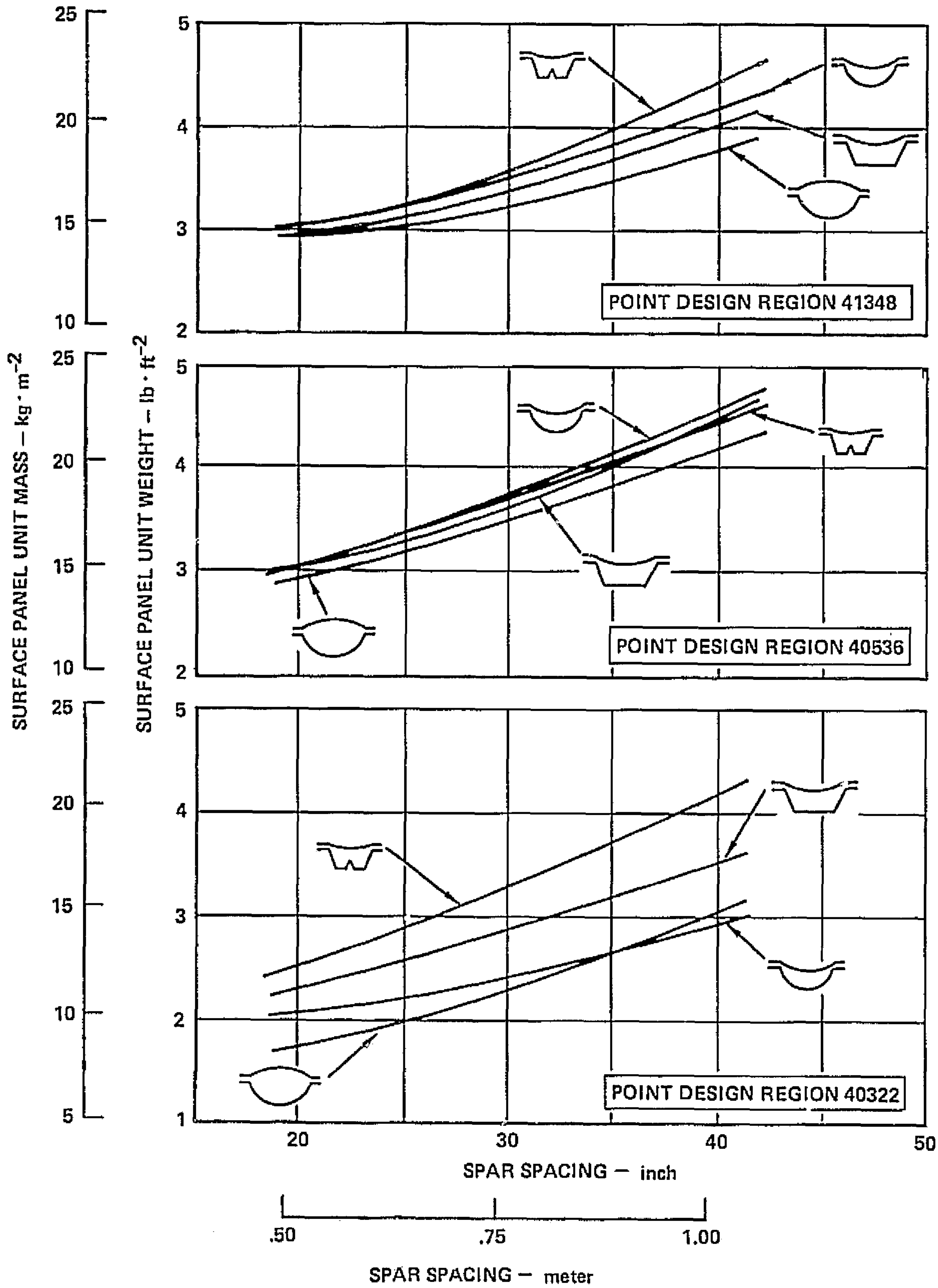


Figure 12-9. Weight Comparison of the Candidate Surface Panel Concepts - Task I Chordwise Wing Arrangement Initial Screening

TABLE 12-9. UNIT WING BOX WEIGHT FOR CHORDWISE PANEL CONCEPTS – TASK I CHORDWISE ARRANGEMENT INITIAL SCREENING

PANEL CONCEPT	CIRCULAR ARC– CONVEX BEADED SKIN			CIRCULAR ARC– CONCAVE BEADED SKIN			TRAPEZOIDAL CORRUGATION– CONCAVE BEADED SKIN			BEADED CORRUGATION CONCAVE BEADED SKIN		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40	20	30	40
PANELS												
UPPER	1.609	1.965	2.241	1.696	2.099	2.620	1.622	1.946	2.435	1.654	2.047	2.419
LOWER	1.335	1.570	1.943	1.367	1.688	1.993	1.461	1.720	2.059	1.422	1.737	2.056
Σ	(2.944)	(3.535)	(4.184)	(3.063)	(3.787)	(4.613)	(3.083)	(3.666)	(4.494)	(3.076)	(3.784)	(4.475)
RIB WEBS												
BULKHEAD	0.238	0.238	0.238	0.238	0.238	0.238	0.238	0.238	0.238	0.238	0.238	0.238
TRUSS	0.228	0.228	0.228	0.228	0.228	0.228	0.228	0.228	0.228	0.228	0.228	0.228
Σ	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)	(0.466)
SPAR WEBS												
BULKHEAD	0.270	0.319	0.375	0.270	0.319	0.375	0.270	0.319	0.375	0.270	0.319	0.375
TRUSS	0.490	0.403	0.325	0.490	0.403	0.325	0.490	0.403	0.325	0.490	0.403	0.325
Σ	(0.760)	(0.722)	(0.700)	(0.760)	(0.722)	(0.700)	(0.760)	(0.722)	(0.700)	(0.760)	(0.722)	(0.700)
RIB CAPS												
UPPER	0.116	0.117	0.130	0.104	0.123	0.152	0.094	0.104	0.122	0.038	0.100	0.109
LOWER	0.086	0.097	0.116	0.087	0.103	0.117	0.093	0.093	0.104	0.087	0.093	0.097
Σ	(0.202)	(0.214)	(0.246)	(0.191)	(0.226)	(0.269)	(0.187)	(0.197)	(0.226)	(0.175)	(0.193)	(0.206)
SPAR CAPS												
UPPER	2.710	2.770	2.890	2.710	2.800	2.930	2.850	2.850	2.950	2.850	2.850	2.950
LOWER	3.950	4.040	4.190	3.950	4.080	4.240	4.010	4.160	4.290	4.010	4.160	4.290
Σ	(6.660)	(6.810)	(7.080)	(6.660)	(6.880)	(7.170)	(6.860)	(7.010)	(7.240)	(6.860)	(7.010)	(7.240)
NON OPTIMUM												
FASTENER	0.200	0.190	0.180	0.200	0.190	0.180	0.200	0.190	0.180	0.200	0.190	0.180
WEB INTERS.	0.120	0.110	0.100	0.120	0.110	0.100	0.120	0.110	0.100	0.120	0.110	0.100
Σ	(0.320)	(0.300)	(0.280)	(0.320)	(0.300)	(0.280)	(0.320)	(0.300)	(0.280)	(0.320)	(0.300)	(0.280)
Σ POINT DESIGN MASS $\left(\frac{lb}{ft^2}\right)$	11.352	12.047	12.956	11.460	12.381	13.498	11.676	12.361	13.406	11.657	12.475	13.367

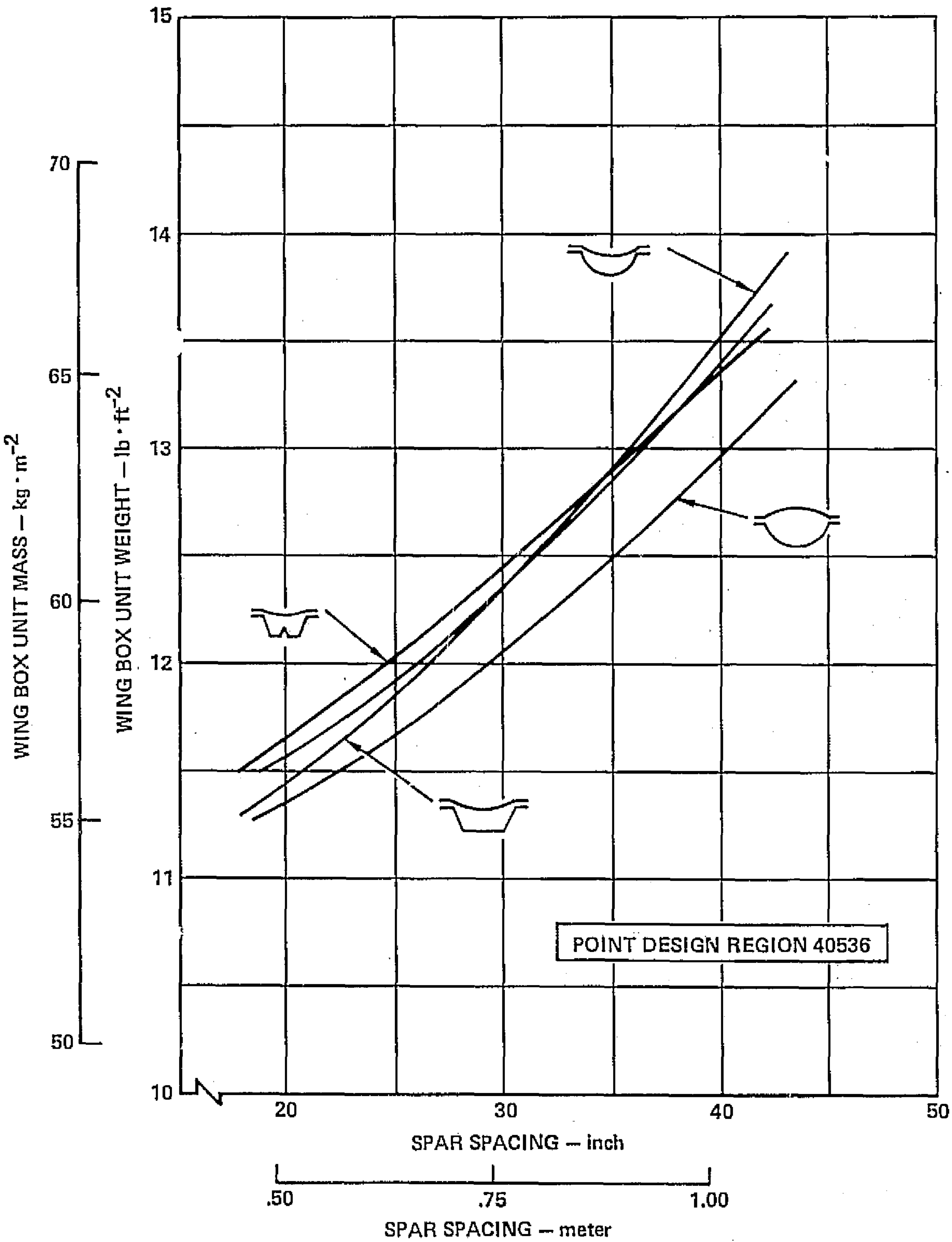


Figure 12-10. Weight Comparison of Chordwise Arrangement Unit Box Weight - Task I Chordwise Initial Screening

11.3 lb/sq.ft. to 13.0 lb/sq.ft. are noted for this concept at spar spacing of 20-inches and 40-inches, respectively.

By comparing the results shown in Figure 12-9 with those in Figure 12-10 it can be seen that the findings of the panel investigation are validated by the results of the wing box analysis, i.e., least-weight concept is the circular-arc convex-beaded I .el.

Chordwise Detailed Concept Analysis

For the detailed concepts analysis, elementary wing boxes, left-hand sketch on Figure 12-6, were subjected to point design weight-strength analyses. Structural components included were the least-weight panel concept surviving the initial screening analysis, circular-arc convex beaded concepts, with substructure commensurate with the chordwise design. This substructure included spar caps and webs, rib caps and webs, and associated non-optimum structure (i.e., posts, shear ties, fasteners, etc.)

Six wing point design regions were selected for this analysis, the three used for the initial screening analysis plus three additional regions. Figure 12-3 contains the locations of these regions. The three additional wing regions were located in the wing aft box and wing tip regions.

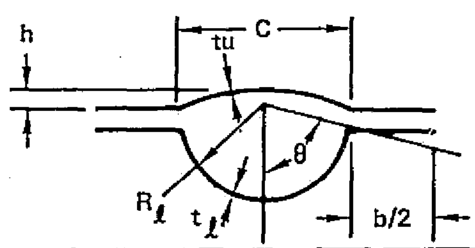
The critical panel load-temperature environments are displayed in Table 12-2 for each of six wing point design regions. The substructure was analysed using the internal forces derived from the NASTRAN redundant structure analysis solution.

Panel Analysis - The results of the panel analysis are summarized in Table 12-10. This table displays the panel dimensions and mass data for the three additional point design regions 40236, 41036, and 41316. All panels were designed for the critical symmetric flight condition at Mach 1.25, Condition 31.

With reference to this table, the bead skin thicknesses ranged from 0.019-inches to 0.072-inches for these regions with no minimum gage restrictions. The radius for the inner bead R_L varied from 0.7-inch to 1.5-inches. The semi-apex angle θ, flat width b, and exterior skin bead height-to-chord ratio h/c were held to th.

TABLE 12-10. PANEL GEOMETRY AND WEIGHT OF THE CIRCULAR ARC-CONVEX BEADED CONCEPT - TASK I CHORDWISE ARRANGEMENT DETAIL CONCEPT ANALYSIS

POINT DESIGN REGION	40236						41036						41316					
SURFACE	UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING (m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																		
t_ℓ (in)	.019	.024	.030	.022	.028	.034	.024	.028	.033	.020	.021	.023	.028	.030	.038	.026	.033	.037
t_u (in)	.019	.024	.028	.025	.030	.033	.030	.037	.040	.030	.029	.030	.072	.072	.067	.051	.046	.050
R_ℓ (in)	0.7	1.0	1.3	0.9	1.2	1.5	0.8	1.1	1.3	0.7	0.8	1.0	0.9	1.1	1.4	0.8	1.0	1.2
θ (deg)	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87
b (in)	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75
MASS DATA:																		
$\bar{t}$ (in)	.045	.058	.071	.056	.070	.082	.063	.077	.087	.057	.058	.062	.112	.115	.122	.087	.092	.103
w (1b/ft^2)	1.032	1.325	1.629	1.279	1.606	1.887	1.452	1.766	2.007	1.320	1.336	1.435	2.571	2.650	2.811	2.007	2.129	2.366
CRITICAL CONDITION	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



PANEL CONCEPT:

CIRCULAR ARC–CONVEX BEADED SKIN (h/c= 0.10)

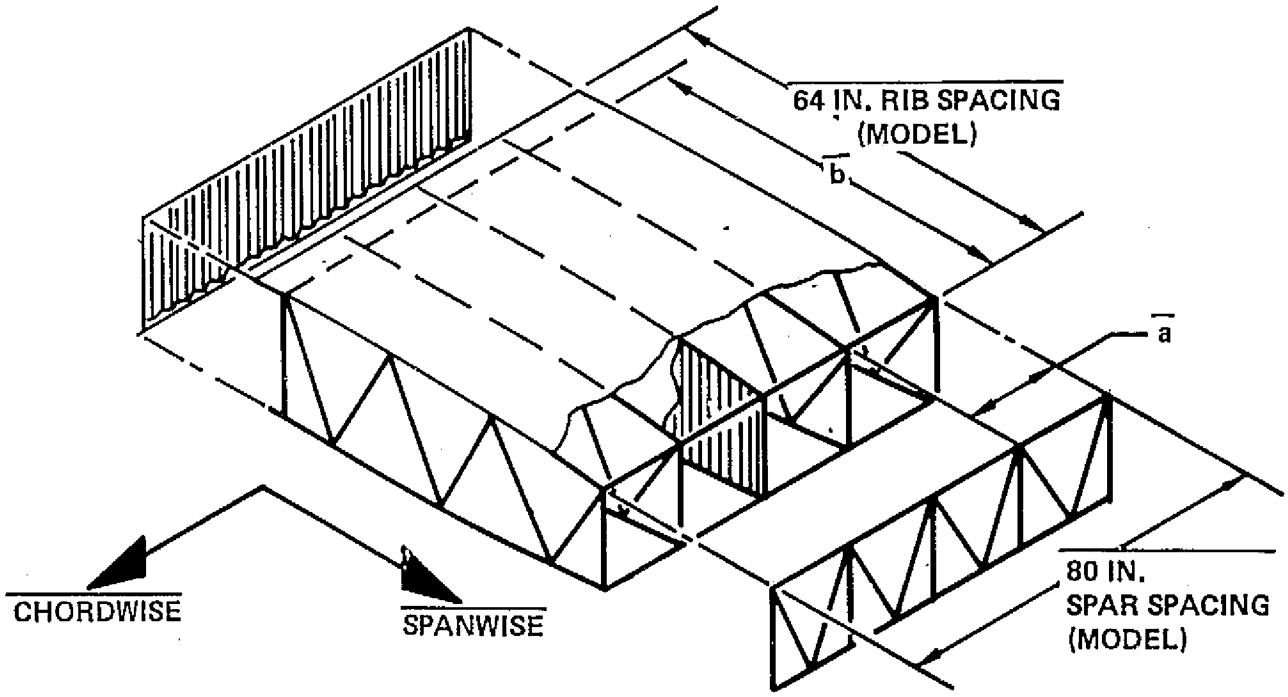
same constraints as the panels during the initial screening analysis: 87-degrees, 0.75-inches, and 0.10, respectively.

The detail concept analysis used the panel weights determined during the initial screening analysis at point design regions 40322, 40536, and 41348. See Table 12-5 for a summary of the panel geometry and weight of these regions.

Substructure Analysis - For the chordwise stiffened wing arrangement, the wing chordwise extensional stiffness and wing torsional stiffness are primarily a function of the surface panel properties with the wing bending stiffness provided by the submerged spar caps. Hence the spars (caps and webs) are major weight components for the chordwise arrangement. In addition, the rib caps and webs, and the associated non-optimum structure were included in this substructure analysis.

The substructure weight-strength analysis was conducted at each of the six point design regions using representative substructure commensurate with the specific region being analyzed, i.e., wet bay or dry bay region. A typical substructure arrangement for point design region 40536 is shown in Figure 12-11. The number of components associated with each rib and spar spacing are shown superimposed on panel dimensions used in the finite-element structural model. All study dimensions were related to the model dimension. To protect the spar caps from the thermal environment and provide clearance for the uses of large surface panels (continuous panel stiffeners) the spar caps are submerged. The caps become large rectangular blocks with integral tee clips attaching to the skin. At bulkheads the surface stiffeners taper out and the tee clips are continuous. At intermediate spars the clips are cut away to allow for continuous panel stiffeners and are hence local discrete elements.

In addition, the model loads reflect spar caps at the wing surface, hence the actual spar cap loads used in the weight-strength analysis were adjusted by the ratio of model spar height. An example of the spar cap stress analysis is shown on Table 12-11, and Table 12-12 presents the resulting spar cap geometry and weight. Study spacings, model spacings and inter-related number of caps are defined for each point design region. The cap geometry and area, and the equivalent surface panel unit weights are displayed for both upper and lower caps. The nomenclature for the cap geometry and the weight equation are shown in the footnotes of this Table. The tension designed lower surface caps (90,000 psi. gross area fatigue allowable stress) are



STUDY SPACING RIB (b) SPAR (a)	 60 20	 60 30	 60 40
SPAR CAPS (NO.) TYPE	4 SLAB	2 2/3 SLAB	2 SLAB
SPAR WEBS (NO.) CORRUGATION TRUSS	4 1 3	2 2/3 1 1 2/3	2 1 1
RIB CAPS (NO.) TYPE	1 TEE	1 TEE	1 TEE
RIB WEBS (NO.) CORRUGATION TRUSS	1 ½ ½	1 ½ ½	1 ½ ½

Figure 12-11. Chordwise Substructure Arrangement Point Design Region 40536

TABLE 12-11. CHORDWISE ARRANGEMENT SPAR CAP ANALYSIS

POINT DESIGN REGION	SURFACE	SPAR SPACING (IN.)	APPLIED LOAD		CAP AREA (IN.^{2})	$f_{T,C}$ (1) (KSi)	$F_{T,C}$ (2) (KSi)	MARGIN (3) OF SAFETY
			COND. NO.	P_{ULT} (KIPS)				
40322	UPPER	20	9	-27.4	0.21	-131.1	-131.0	0.00
		30	9	-41.6	0.32	-130.8	-131.0	0.00
		40	9	-56.0	0.43	-129.9	-131.0	0.01
	LOWER	20	9	27.4	0.30	90.1	90.0	0.00
		30	9	41.6	0.46	89.8	90.0	0.00
		40	9	56.0	0.62	90.3	90.0	0.00
40536	UPPER	20	31	-307.4	2.35	-130.8	-131.0	0.00
		30	31	-472.8	3.62	-130.6	-131.0	0.00
		40	31	-656.6	5.02	-130.8	-131.0	0.00
	LOWER	20	31	307.4	3.43	89.6	90.0	0.00
		30	31	472.8	5.27	89.7	90.0	0.00
		20	31	656.6	7.27	90.3	90.0	0.00
41348	UPPER	20	31	-272.2	2.09	-130.2	-131.0	0.01
		30	31	-436.0	3.33	-130.9	-131.0	0.00
		40	31	-632.0	4.83	-130.8	-131.0	0.00
	LOWER	20	31	272.2	3.02	90.1	90.0	0.00
		30	31	436.0	4.83	90.3	90.0	0.00
		40	31	632.0	7.05	89.6	90.0	0.00

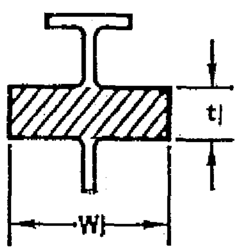
NOTES

1. APPLIED STRESS = P_{ULT}/A
2. ALLOWABLE STRESS:
   COMPRESSION (F_C) = F_{CY}
   TENSION (F_T) = 90,000 PSI (FATIGUE ALLOWABLE)
3. MARGIN OF SAFETY = F/f - 1

TABLE 12-12. SPAR CAP GEOMETRY AND WEIGHT OF THE CHORDWISE WING ARRANGEMENT

POINT DESIGN REGION	SPACING		MODEL SPACING		N NUMBER SPARS	UPPER SURFACE SPAR CAPS				LOWER SURFACE SPAR CAPS			
	SPAR a (IN.)	RIB b (IN.)	SPAR A (IN.)	RIB B (IN.)		W (IN.)	t (IN.)	A_C (IN.^{2})	w (LB/SQ. FT)	W (IN.)	t (IN.)	A_C (IN.^{2})	w (LB/SQ. FT)
40322	20	60	90	71	4.50	1.50	.139	.209	.241	1.50	.203	.304	.350
	30	60	90	71	3.00	1.50	.212	.318	.244	1.50	.308	.463	.356
	40	60	90	71	2.25	1.50	.287	.431	.248	1.50	.413	.620	.357
40236	20	60	80	63	4.00	2.0	1.37	2.74	3.16	3.0	1.37	4.12	4.75
	30	60	80	63	2.66	3.0	1.41	4.23	3.25	3.5	1.81	6.34	4.86
	40	60	80	63	2.00	3.0	1.92	5.76	3.31	4.0	2.16	8.64	4.97
40536	20	60	80	64	4.00	2.0	1.18	2.35	2.71	2.5	1.37	3.43	3.95
	30	60	80	64	2.66	2.5	1.45	3.62	2.77	3.0	1.76	5.27	4.04
	40	60	80	64	2.00	3.0	1.67	5.02	2.89	3.5	2.08	7.27	4.19
41036	20	60	100	64	5.00	2.0	0.810	1.62	1.87	2.0	1.17	2.34	2.71
	30	60	100	64	3.33	2.5	1.03	2.58	1.98	2.5	1.50	3.75	2.88
	40	60	100	64	2.50	3.0	1.21	3.63	2.08	3.0	1.75	5.25	3.02
41316	20	60	50	45	2.5	2.5	1.36	3.40	3.92	3.0	1.66	4.98	5.73
	30	60	50	45	1.66	3.0	1.81	5.43	4.15	3.5	2.25	7.88	6.04
	40	60	50	45	1.25	3.5	2.23	7.81	4.50	4.0	2.84	11.36	6.55
41348	20	60	35	40	1.75	2.0	1.045	2.09	2.41	2.5	1.208	3.02	3.48
	30	60	35	40	1.17	2.5	1.332	3.33	2.56	3.0	1.610	4.83	3.72
	40	60	35	40	0.875	3.0	1.610	4.83	2.78	3.5	2.014	7.05	4.06

SPAR CAP DIMENSIONS



A_C = CAP AREA OF EFFECTIVE LOAD CARRYING MATERIAL
$= W \times t$

w = EQUIVALENT SURFACE PANEL WEIGHT, LB/SQ. FT.

$$= \frac{N \times A_C}{A} \times 23.04$$

heavier than the compression designed upper surface caps. At point design region 40236, inboard location on aft wing box, the lower surface caps are approximately 1.6 lb/sq.ft. heavier than the upper surface caps. As to be expected, the spanwise cap areas increased at the wing tip root and the wing/fuselage intersection. On the aft box, reading from outboard to inboard, the unit weights of the upper surface spar caps for 20.0-inch spacing ranged from approximately 1.90 lb/sq.ft. to 3.2 lb/sq.ft., respectively. The corresponding weights of the lower surface caps ranged from approximately 2.70 lb/sq.ft. to 4.8 lb/sq.ft. A general ranking of the wing regions by their spar cap weights are as follows: heaviest caps are indicated for the wing tip regions (41316 and 41348), intermediate weights for the aft box regions (40236, 40536, and 41036), and least-weight for the lightly loaded spars in the forward wing box region 40322.

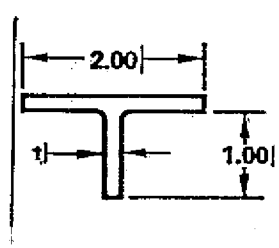
The rib caps carry relatively low loads as the chordwise stiffened surface panels are the primary load-carrying structure. The rib caps were designed to support the surface panels, provided the necessary material for panel splicing, and in association with the rib web, supply additional chordwise bending stiffness.

The rib cap geometry was held constant, a tee 2.0-inches wide with a 1.0-inch flange, with only the thickness allowed to vary with the load intensity. A minimum design restriction of 0.04-inches was imposed on the thickness. A summary of the rib cap geometry and weight are shown in Table 12-13. This table contains the same type of spacing data and number of rib caps as described for the spar caps analysis. The upper and lower surface rib cap geometry for each region are displayed along with the equivalent surface panel unit weight. The footnotes contain the typical rib cap geometry, and the cap area and equivalent panel unit weight equations. The rib caps at point design region 41316 (wing tip/aft box interface) required the greatest areas (heaviest weight) with the caps at region 40322 being the least-weight designs. A unit weight of 0.16 lb/sq.ft. is noted for the upper rib cap for 20-inch spar spacing at region 41316 and 0.058 lb/sq.ft. for the corresponding spacing design at region 40322.

A combination of circular-arc corrugation and truss designs was considered for the spar and rib webs. The type of web used was contingent on the location of the specific point design region being analyzed. For wet-bay regions, a combination of bulkhead and truss webs was considered; whereas, for the dry-bay wing tip regions the relatively small wing thickness prohibited the use of truss webs and

TABLE 12-13. RIB CAP GEOMETRY AND WEIGHT OF THE CHORDWISE WING ARRANGEMENT

POINT DESIGN REGION	SPACING		MODEL SPACING		NUMBER RIB CAPS N	UPPER RIB CAP			LOWER RIB CAP		
	SPAR a (IN.)	RIB b (IN.)	SPAR A (IN.)	RIB B (IN.)		t (IN.)	A_R (IN.^{2})	w (LB/SQ. FT)	t (IN.)	A_R (IN.^{2})	w (LB/SQ. FT)
40322	20	60	90	71	1.50	.040	.120	.058	.045	.135	.065
	30	60	90	71	1.50	.055	.166	.081	.050	.150	.073
	40	60	90	71	1.50	.066	.199	.097	.050	.150	.073
40236	20	60	80	63	1.00	.064	.191	.070	.076	.228	.083
	30	60	80	63	1.00	.077	.230	.084	.090	.271	.099
	40	60	80	63	1.00	.090	.270	.099	.102	.306	.112
40536	20	60	80	64	1.00	.107	.321	.116	.080	.240	.086
	30	60	80	64	1.00	.108	.324	.117	.090	.270	.097
	40	60	80	64	1.00	.120	.360	.130	.107	.321	.116
41036	20	60	100	64	1.66	.052	.156	.093	.049	.147	.087
	30	60	100	64	1.66	.061	.182	.109	.049	.147	.087
	40	60	100	64	1.66	.067	.201	.120	.051	.153	.091
41316	20	60	50	45	.750	.139	.416	.160	.109	.327	.126
	30	60	50	45	.750	.141	.423	.162	.112	.336	.129
	40	60	50	45	.750	.145	.435	.167	.122	.366	.141
41348	20	60	35	40	0.666	.089	.267	.103	.064	.192	.074
	30	60	35	40	0.666	.101	.303	.116	.076	.228	.087
	40	60	35	40	0.666	.112	.336	.129	.077	.231	.088



A = AREA, IN.^{2}

= 3.00 x t

w = EQUIVALENT SURFACE PANEL UNIT WEIGHT, LB/FT.^{2}

$$= \frac{N \times A_R \times 23.04}{B}$$

only corrugated webs were considered. The spar web geometry and corresponding weights are summarized in Table 12-14, with the similar data for the rib webs shown in Table 12-15. Both tables contain the pertinent type and number of webs for each point design region. In addition, the geometry nomenclature, and the area and weight equations are defined in the footnotes.

Unit Box Weights - Tables 12-16 and 12-17 contain a summary of the component and total weights of the chordwise wing concept for the six point design regions. The components included: the upper and lower surface panels, rib webs, rib caps, spar webs, spar caps, and non-optimum factors. In addition to the component weights, the total point design box weight is also included on these tables. For ease in interpreting these results, the component and total weights at each point design region as a function of spar spacing are presented in graphic form in Figure 12-12 through 12-17.

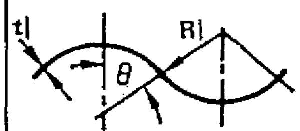
The results for the lightly loaded forward wing box region 40322 are presented in Figure 12-12. The total box weight curve has a positive slope of approximately .025 lb./sq.ft. per inch of spacing with a minimum-weight value of 3.8 lb/sq.ft occurring at the 20-inch spar spacing design. With respect to the panel weight curves the compression design upper surface panels are heavier than the lower surface panels for all designs with approximately 25-inch or larger spar spacings. The panel weights for the 25-inch spar spacing designs are 1.05 lb/sq.ft. For the 20-inch spar design the incremental weight between the upper and lower surface is approximately 0.12 lb/sq.ft. with the upper surface design (0.82 lb/sq.ft.) being the lightest. All substructure components are less than 1.0 lb/sq.ft. with the spar webs for the 20-inch spacing design being the heaviest weight component.

The wing box weights for region 40236 are shown in Figure 12-13. For this chordwise stiffened wing panel concept, the wing spanwise bending loads are carried by submerged spar caps which for this region are the heaviest component and weight approximately 8.0 lb/sq.ft. The surface panels are relatively light-weight components with the heaviest panels being the lower surface panels which weigh approximately 1.30 lb/sq.ft. and 1.90 lb/sq.ft. for the 20-inch and 40-inch spar spacing designs, respectively. Of the remaining components, the spar webs were the heaviest item at 0.90 lb/sq.ft. for the 20-inch spar spacing design. The total weight curve is linear with a positive slope, a minimum total weight of approximately 12.0 lb/sq.ft. occurs for the 20-inch spar spacing design.

TABLE 12-14. SPAR WEB GEOMETRY AND WEIGHT OF THE CHORDWISE WING ARRANGEMENT

POINT DESIGN REGION	SPACING: SPAR a (IN.)	SPACING: RIB b (IN.)	MODEL SPACING: SPAR A (IN.)	MODEL SPACING: RIB B (IN.)	N NUMBER BULKHD WEBS	BULKHEAD WEBS: h (IN.)	BULKHEAD WEBS: θ (DEG.)	BULKHEAD WEBS: R (IN.)	BULKHEAD WEBS: t (IN.)	BULKHEAD WEBS: w (LB/SQ. FT)	N NUMBER TRUSS WEBS	TRUSS WEBS: TUBE DIA. D (IN.)	TRUSS WEBS: WALL THK. t (IN.)	TRUSS WEBS: L (IN.)	TRUSS WEBS: A ($IN.^2$)	TRUSS WEBS: NUMBER DIAGONALS n	TRUSS WEBS: w (LB/SQ. FT)
40322	20	60	90	71	1	42	80	1.20	.022	.336	3.5	1.50	.023	44.6	.107	5	.301
	30	60	90	71	1	42	79	1.30	.022	.333	2.0	1.50	.026	44.6	.119	5	.191
	40	60	90	71	1	42	68	1.60	.026	.336	1.25	1.50	.033	44.6	.152	5	.153
40236	20	60	80	63	1	35	63	1.80	.029	.361	3.0	2.25	.037	38.0	.261	4	.544
	30	60	80	63	1	35	79	1.40	.028	.396	1.66	2.25	.053	38.0	.365	4	.421
	40	60	80	63	1	35	75	1.50	.033	.451	1.00	2.25	.068	38.0	.465	4	.323
40536	20	60	80	64	1	27	67	1.40	.027	.270	3.0	1.625	.065	28.3	.321	4	.490
	30	60	80	64	1	27	72	1.30	.031	.319	1.66	1.625	.099	28.3	.475	4	.403
	40	60	80	64	1	27	63	1.70	.039	.375	1.00	1.625	.136	28.3	.636	4	.325
41036	20	60	100	64	1	16.5	62	1.00	.023	.109	4.0	1.50	.063	22.0	.284	4	.359
	30	60	100	64	1	16.5	63	1.00	.028	.132	2.33	1.50	.063	22.0	.389	4	.287
	40	60	100	64	1	16.5	65	1.00	.032	.151	1.50	1.50	.096	22.0	.423	4	.201
41316	20	60	50	45	2.5	13.5	60	1.00	.023	.439	–	–	–	–	–	–	–
	30	60	50	45	1.66	13.5	60	1.00	.028	.354							
	40	60	50	45	1.25	13.5	60	1.00	.031	.288							
41348	20	60	35	40	1.75	11.0	60	1.00	.019	.291	–	–	–	–	–	–	–
	30	60	35	40	1.17	11.0	70	1.00	.024	.264							
	40	60	35	40	0.875	11.0	60	1.00	.025	.192							

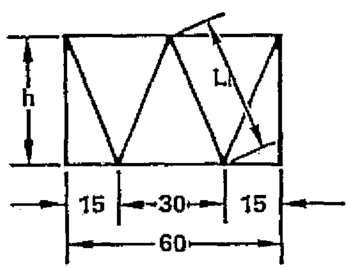
BULKHEAD WEBS (CIRCULAR-ARC CORRUGATION)



w = EQUIV. SURFACE PANEL WT., LB/SQ. FT.

$$= N\left(\frac{\theta}{\sin\theta}\right)\left(\frac{h}{A}\right) t \times 23.04$$

TRUSS WEBS (DIAGONAL ELEMENTS)



A_D = DIAGONAL AREA

$$= \pi/4\left(D_O^2 - D_L^2\right)$$

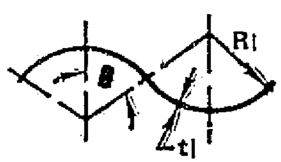
w = EQUIV. SURFACE PANEL WT., LB/SQ. FT.

$$= \frac{N(n \times A_D \times L)}{A \times B} \times 23.04$$

TABLE 12-15. RIB WEB GEOMETRY AND WEIGHT OF THE CHORDWISE WING ARRANGEMENT

POINT DESIGN REGION	SPACING: SPAR a (IN.)	SPACING: RIB b (IN.)	MODEL SPACING: SPAR A (IN.)	MODEL SPACING: RIB B (IN.)	N NUMBER BULKHD WEBS	BULKHEAD WEBS: h (IN.)	BULKHEAD WEBS: θ (DEG.)	BULKHEAD WEBS: R (IN.)	BULKHEAD WEBS: t (IN.)	BULKHEAD WEBS: w (LB/SQ. FT.)	N NUMBER TRUSS WEBS	TRUSS WEBS: TUBE DIA. D (IN.)	TRUSS WEBS: WALL THK. t (IN.)	TRUSS WEBS: L (IN.)	TRUSS WEBS: A_D (IN.^{2})	TRUSS WEBS: n NUMBER DIAGONALS	TRUSS WEBS: w (LB/SQ. FT.)
	20	60	90	71	0.50	45	78	1.50	.029	.298	1.00	1.50	.021	47.4	.097	4.5	.074
40322	30	60	90	71	0.50	45	78	1.50	.029	.298	1.00	1.50	.016	45.5	.074	6.0	.074
	40	60	90	71	0.50	45	78	1.50	.029	.298	1.00	1.50	.021	47.4	.097	4.5	.074
	20	60	80	63	0.50	36	65	1.80	.034	.279	0.50	2.25	.093	41.0	.630	4.0	.237
40236	30	60	80	63	0.50	36	65	1.80	.034	.279	0.50	2.25	.073	39.0	.499	5.33	.237
	40	60	80	63	0.50	36	65	1.80	.034	.279	0.50	2.25	.093	41.0	.630	4.00	.237
	20	60	80	64	0.50	27	67	1.50	.038	.238	0.50	2.00	.128	33.6	.753	4.0	.228
40536	30	60	80	64	0.50	27	67	1.50	.038	.238	0.50	2.0	.103	30.9	.614	5.33	.228
	40	60	80	64	0.50	27	67	1.50	.038	.238	0.50	2.00	.128	33.6	.753	4.00	.228
	20	60	100	64	1.00	16	60	1.00	.016	.111	0.666	1.00	.065	26.0	.192	5.00	.060
41036	30	60	100	64	1.00	16	60	1.00	.016	.111	0.666	1.00	.058	22.0	.172	6.66	.060
	40	60	100	64	1.00	16	60	1.00	.016	.111	0.666	1.00	.065	26.0	.192	5.00	.060
	20	60	50	45	0.75	13	60	1.00	.045	.270	–	–	–	–	–	–	–
41316	30	60	50	45	0.75	13	60	1.00	.045	.270	–	–	–	–	–	–	–
	40	60	50	45	0.75	13	60	1.00	.045	.270							
	20	60	35	40	0.666	12	60	1.00	.019	.106							
41348	30	60	35	40	0.666	12	60	1.00	.019	.106	–	–	–	–	–	–	–
	40	60	35	40	0.666	12	60	1.00	.019	.106							

BULKHEAD WEBS (CIRCULAR-ARC CORRUGATION)



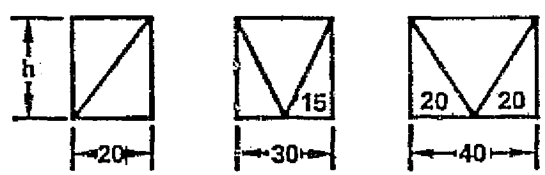
h = AVERAGE WEB HEIGHT

w = EQUIVALENT SURFACE PANEL WEIGHT

$$= N \left(\frac{\theta}{\sin\theta}\right)\left(\frac{h}{B}\right) t \times 23.04;$$

θ = RADIANS

TRUSS WEBS (DIAGONAL ELEMENTS)



SPAR SPACING

A_D = DIAGONAL AREA

$$= \frac{\pi}{4}\left(D_o^2 - D_i^2\right)$$

$$w = \frac{N(n \times A_D \times L)}{A \times B} \times 23.04$$

TABLE 12-16. DETAIL WING WEIGHTS FOR THE CHORDWISE WING ARRANGEMENT

POINT DESIGN REGION	40322			40536			41348		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	0.825	1.263	1.619	1.609	1.965	2.241	1.632	1.925	2.199
LOWER	0.942	1.120	1.413	1.335	1.570	1.943	1.366	1.328	1.617
Σ	(1.767)	(2.383)	(3.032)	(2.944)	(3.535)	(4.184)	(2.998)	(3.253)	(3.816)
RIB WEBS									
BULKHEAD	0.298	0.298	0.298	0.238	0.238	0.238	0.106	0.103	0.106
TRUSS	0.074	0.074	0.074	0.228	0.228	0.228	–	–	–
Σ	(0.372)	(0.372)	(0.372)	(0.466)	(0.466)	(0.466)	(0.106)	(0.106)	(0.106)
SPAR WEBS									
BULKHEAD	0.336	0.333	0.336	0.270	0.319	0.375	0.291	0.264	0.192
TRUSS	0.301	0.191	0.153	0.490	0.403	0.325	–	–	–
Σ	(0.637)	(0.524)	(0.489)	(0.760)	(0.722)	(0.700)	(0.291)	(0.264)	(0.192)
RIB CAPS									
UPPER	0.058	0.081	0.097	0.116	0.117	0.130	0.103	0.116	0.129
LOWER	0.065	0.073	0.073	0.086	0.097	0.116	0.074	0.087	0.088
Σ	(0.123)	(0.154)	(0.170)	(0.202)	(0.214)	(0.246)	(0.177)	(0.203)	(0.217)
SPAR CAPS									
UPPER	0.241	0.244	0.248	2.710	2.770	2.890	2.410	2.560	2.780
LOWER	0.350	0.356	0.357	3.950	4.040	4.190	3.480	3.720	4.060
Σ	(0.591)	(0.600)	(0.605)	(6.660)	(6.810)	(7.080)	(5.890)	(6.280)	(6.840)
NON-OPTIMUM									
MECH. FAST.	0.180	0.170	0.160	0.200	0.190	0.180	0.200	0.190	0.180
WEB INTERS.	0.120	0.110	0.100	0.120	0.110	0.100	0.120	0.110	0.100
Σ	(0.300)	(0.280)	(0.260)	(0.320)	(0.300)	(0.280)	(0.320)	(0.300)	(0.280)
Σ POINT DESIGN MASS $\frac{LB}{FT^2}$	3.790	4.313	4.928	11.352	12.047	12.956	9.782	10.406	11.451

TABLE 12-17. DETAIL WING WEIGHTS FOR THE CHORDWISE WING ARRANGEMENT

POINT DESIGN REGION	40236			41036			41316		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	1.032	1.325	1.629	1.452	1.764	2.007	2.571	2.650	2.811
LOWER	1.279	1.606	1.887	1.320	1.336	1.435	2.007	2.129	2.366
Σ	(2.311)	(2.931)	(3.516)	(2.772)	(3.100)	(3.442)	(4.578)	(4.779)	(5.177)
RIB WEBS									
BULKHEAD	0.279	0.279	0.279	0.111	0.111	0.111	0.270	0.270	0.270
TRUSS	0.237	0.237	0.237	0.060	0.060	0.060	–	–	–
Σ	(0.516)	(0.516)	(0.516)	(0.171)	(0.171)	(0.171)	(0.270)	(0.270)	(0.270)
SPAR WEBS									
BULKHEAD	0.361	0.396	0.451	0.109	0.132	0.151	0.439	0.354	0.288
TRUSS	0.544	0.421	0.323	0.359	0.287	0.201	–	–	–
Σ	(0.905)	(0.817)	(0.774)	(0.468)	(0.419)	(0.352)	(0.439)	(0.354)	(0.288)
RIB CAPS									
UPPER	0.070	0.084	0.099	0.093	0.109	0.120	0.160	0.162	0.167
LOWER	0.083	0.099	0.112	0.087	0.087	0.091	0.126	0.129	0.141
Σ	(0.153)	(0.183)	(0.211)	(0.180)	(0.196)	(0.211)	(0.286)	(0.291)	(0.308)
SPAR CAPS									
UPPER	3.160	3.250	3.310	1.870	1.980	2.080	3.920	4.150	4.500
LOWER	4.750	4.860	4.970	2.710	2.880	3.020	5.730	6.040	6.550
Σ	(7.910)	(8.110)	(8.280)	(4.580)	(4.860)	(5.100)	(9.650)	(10.190)	(11.050)
NON-OPTIMUM									
MECH. FAST.	0.200	0.190	0.180	0.200	0.190	0.180	0.200	0.190	0.180
WEB INTERS.	0.120	0.110	0.100	0.120	0.110	0.100	0.120	0.110	0.100
Σ	(0.320)	(0.300)	(0.280)	(0.320)	(0.300)	(0.280)	(0.320)	(0.300)	(0.280)
Σ POINT DESIGN MASS $\frac{LB}{FT^2}$	12.115	12.057	12.577	8.491	9.046	9.556	15.543	16.184	17.373

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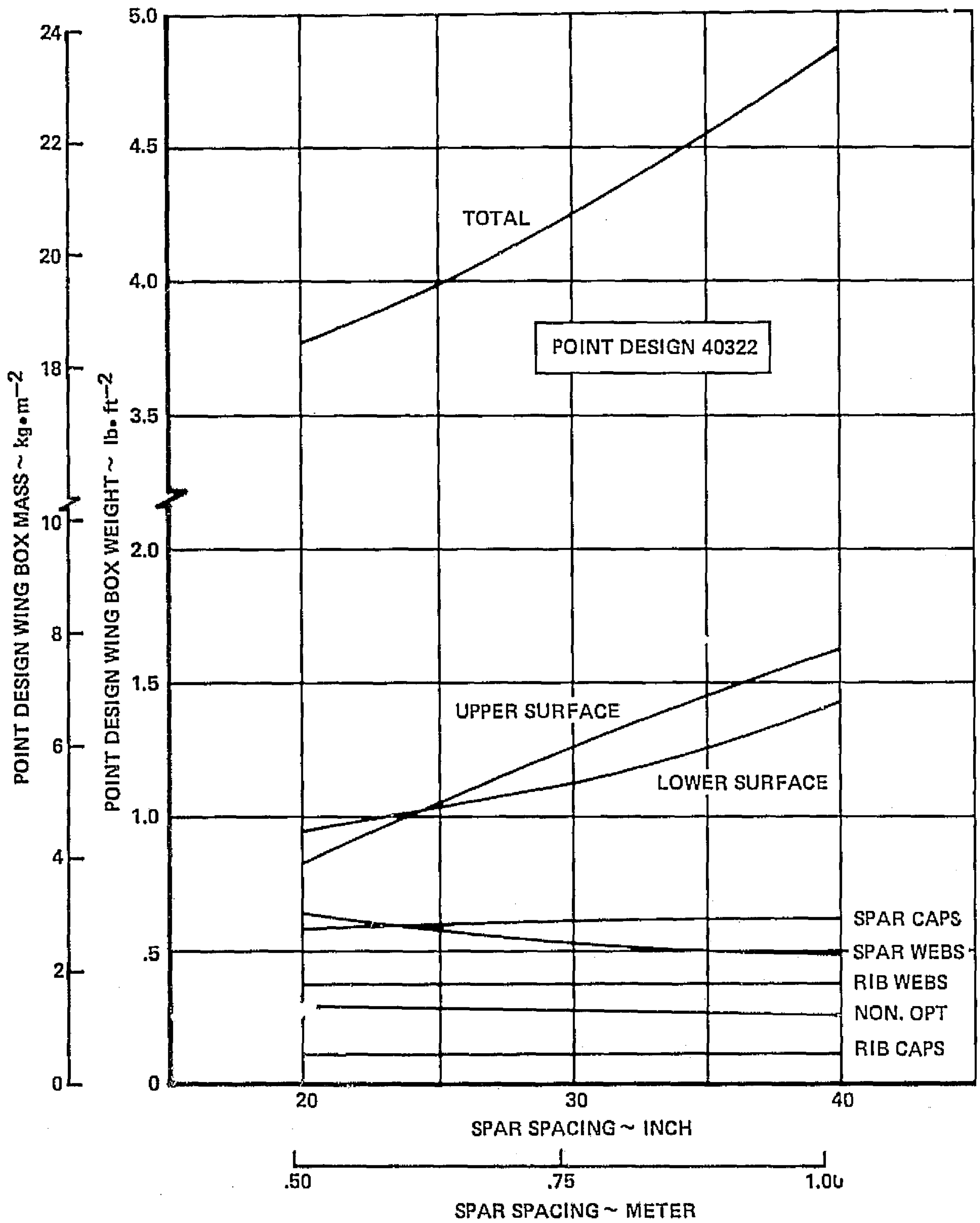


Figure 12-12. Optimum Spar Spacing for Chordwise Wing Arrangement at Point Design Region 40322

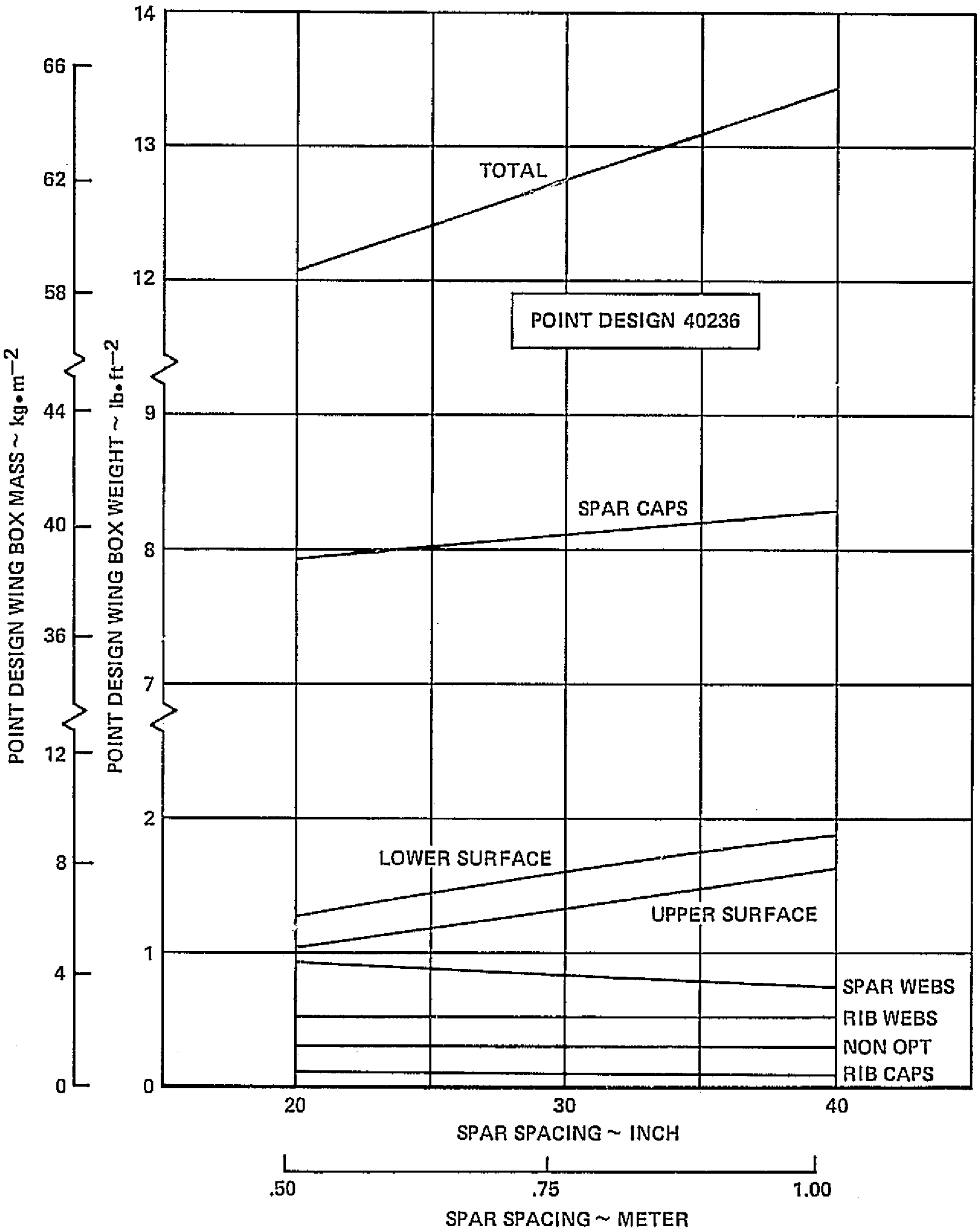


Figure 12-13. Optimum Spar Spacing for Chordwise Wing Arrangement at Point Design Region 40236

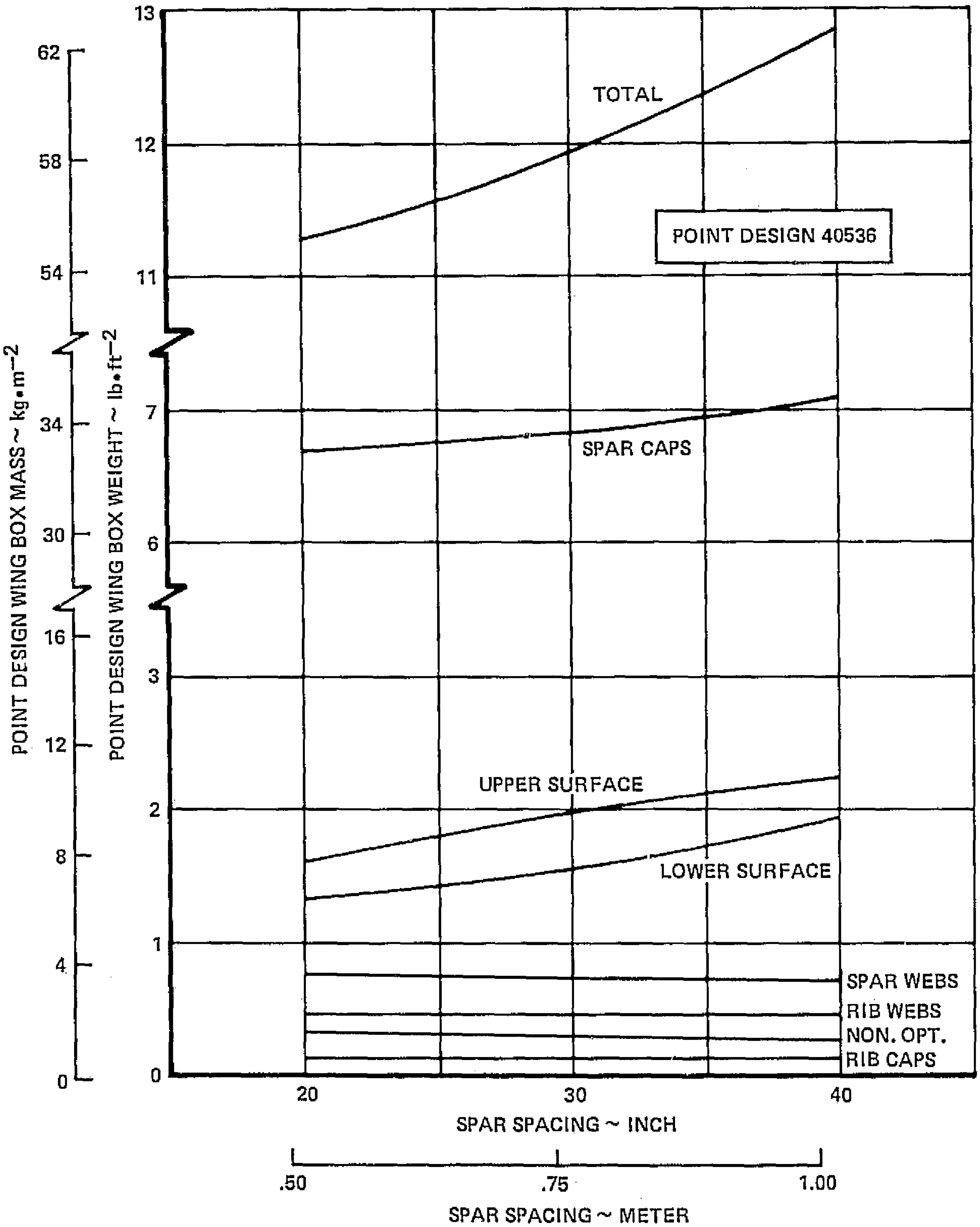


Figure 12-14. Optimum Spar Spacing for Chordwise Wing Arrangement at Point Design Region 40536

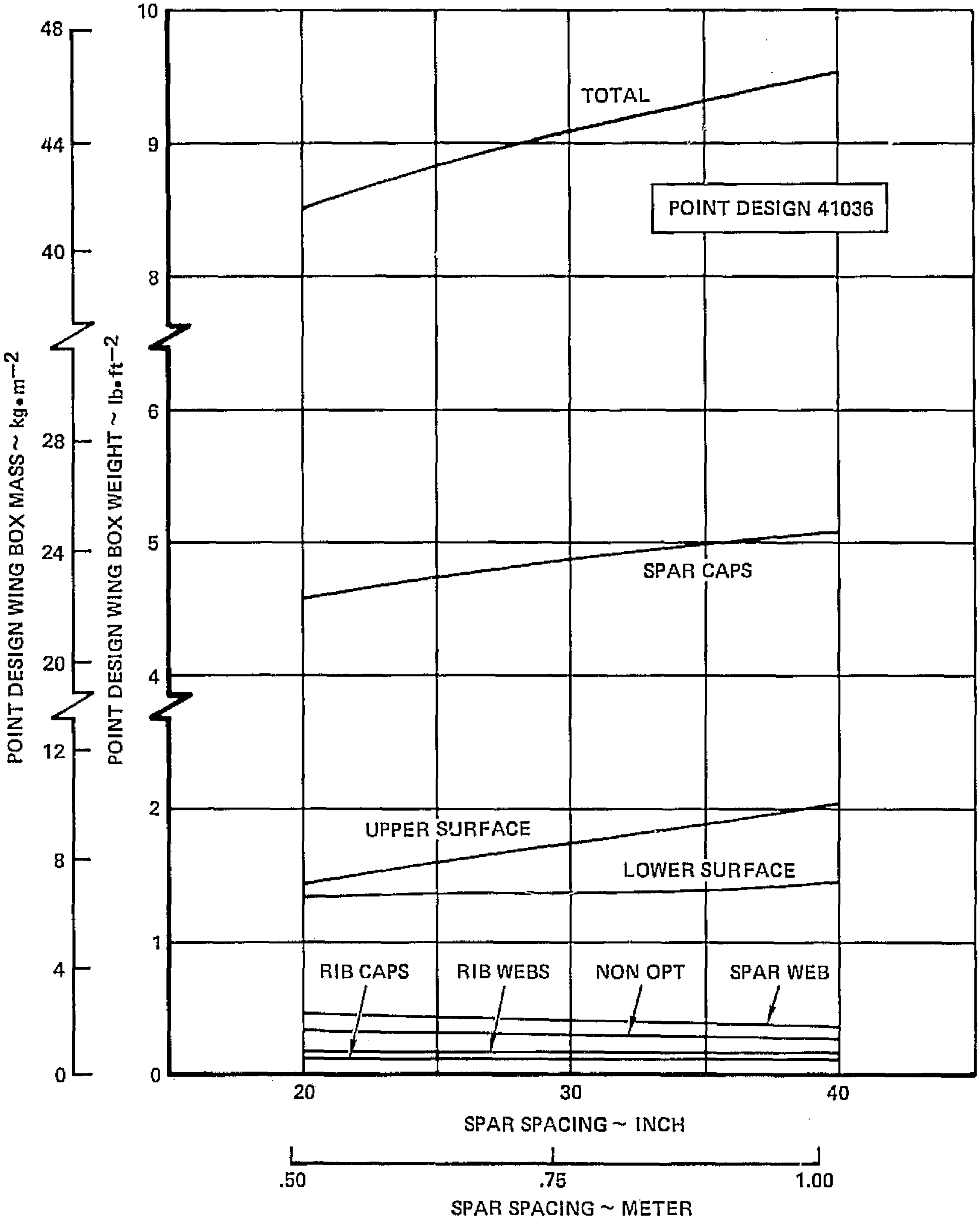


Figure 12-15. Optimum Spar Spacing for Chordwise Wing Arrangement at Point Design Region 41036

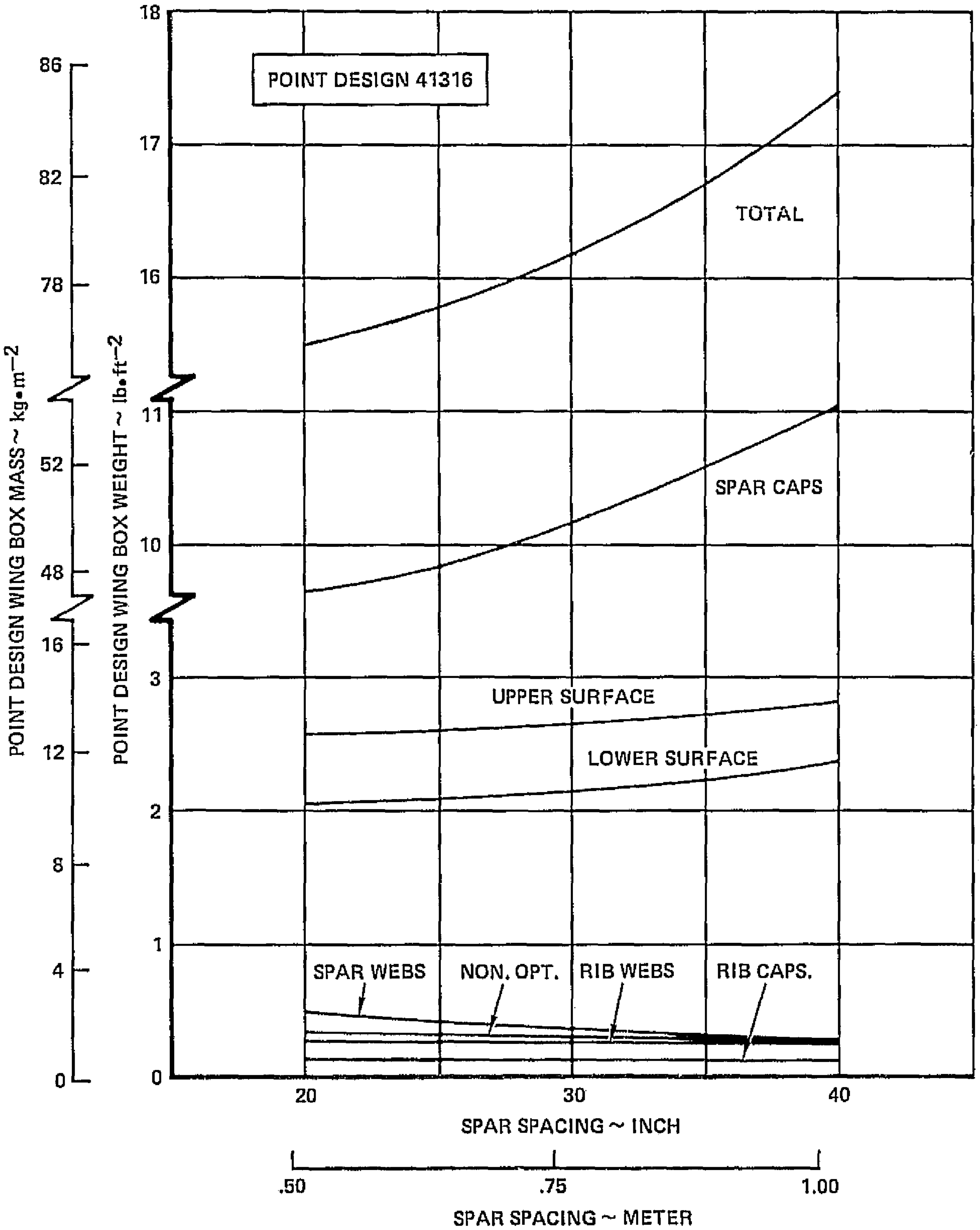


Figure 12-16. Optimum Spar Spacing for Chordwise Wing Arrangement at Point Design Region 41316

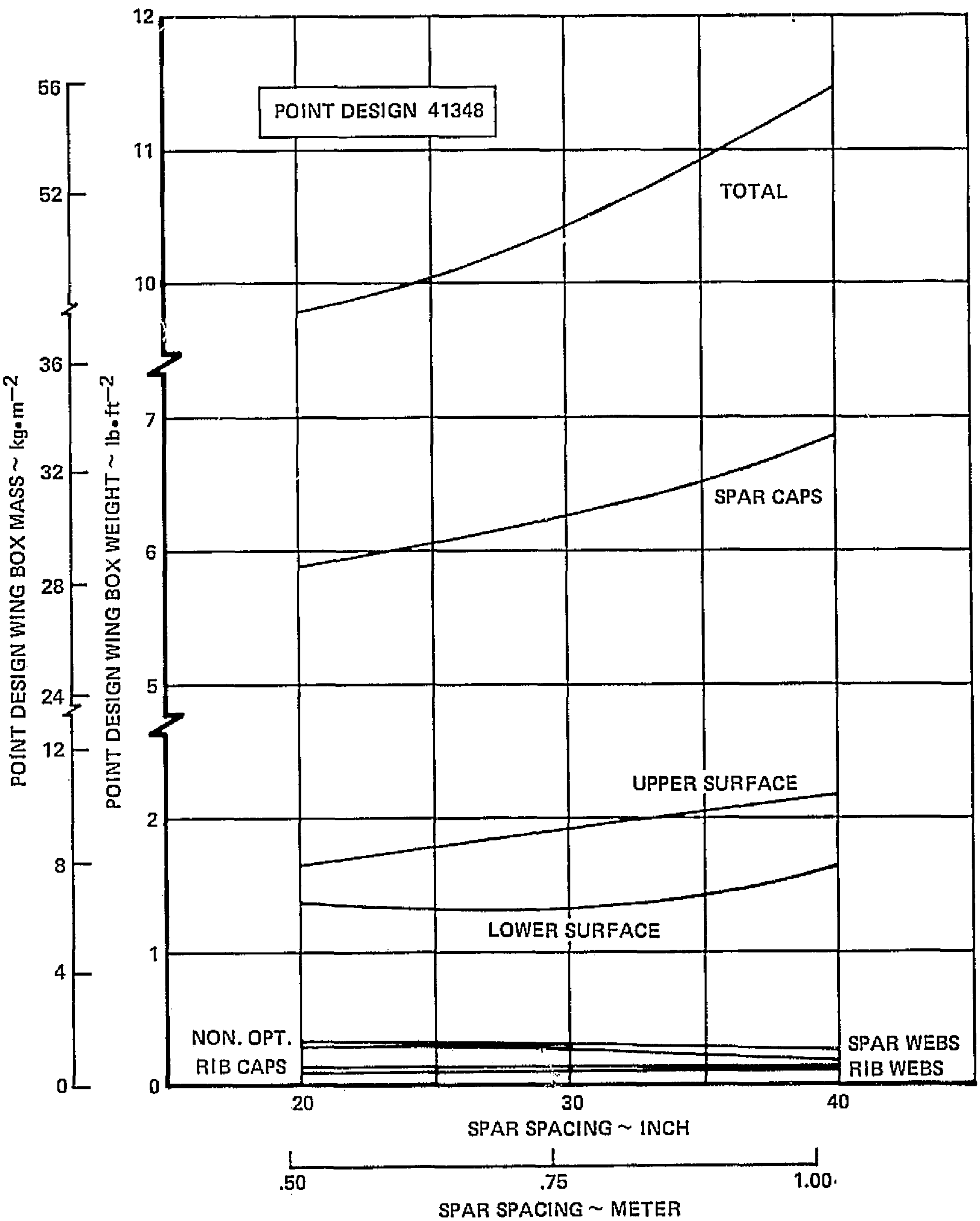


Figure 12-17. Optimum Spar Spacing for Chordwise Wing Arrangement at Point Design Region 41348

The results of the weight-strength analysis for region 40536 are presented in Figure 12-14 and indicate the least weight design (11.4 lb/sq.ft.) occurs at the smallest spar spacing investigated, 20-inches. As with region 40236, the spar caps (upper and lower caps) were the heaviest weight components ranging from 6.7 lb/sq.ft. at the 20-inch spar spacing to 7.1 lb/sq.ft. at the 40-inch spar spacing. The spar cap weight amounts to approximately 55- to 59-percent of the total box weight with the larger percentage occurring for the 20-inch spar spacing design. The surface panel weight, combined weight of the upper and lower panels, varied from 2.9 lb/sq.ft. to 4.2 lb/sq.ft. for the 20-inch and 40-inch spar spacing designs, respectively, with the upper surface panel being the heaviest panel for all designs. The surface panel weight amounts to approximately 26- to 32-percent of the total box weight. The sum of the weight of the remaining structural components (spar and rib webs, rib caps, and non-optimum factor) amount to a maximum of 15-percent of the total weight for the 20-inch spar spacing design and approximately 13-percent for the 40-inch spar spacing design.

The wing box weights for region 41036 are presented in Figure 12-15. As with the previous discussed regions the least-weight design corresponded to the smallest spar spacing (20-inches) design investigated and weighed approximately 8.5 lb/sq.ft. The upper surface panel was the heaviest panel for all spar spacings investigated with the weight ranging from 1.45 lb/sq.ft. to 2.00 lb/sq.ft. for the 20-inch and 40-inch spar spacing design, repectively. The spar caps (upper and lower) weighed approximately 5.0 lb/sq.ft. and amounted to approximately 54-percent of the total box weight. With reference to the remaining structure, no single component weighed more than 0.5 lb/sq.ft.

The wing box weights for the wing tip inboard point design region, region 41316, are presented in Figure 12-16. The minimum-weight design (15.5 lb/sq.ft.) occurs at the 20-inch spar spacing with the 40-inch spar spacing design being the heaviest at 17.4 lb/sq.ft. For the minimum-weight design, the weight of the spar caps amounts to 62-percent of the total weight, while the weight attributed to the panels, upper and lower, accounts for approximately 29-percent of the total weight. The remaining structure for the 20-inch spar spacing design amounts to approximately 9-percent of the total weight. The corresponding weights of these components for the least-weight 20-inch design are: 9.65 lb/sq.ft., 4.58 lb/sq.ft., and 1.31 lb/sq.ft., respectively.

The component and total box weight for region 41348 are presented in Figure 12-17. The minimum-weight design is the 20-inch spar spacing design which has a total box weight of 9.8 lb/sq.ft., with a unit weight of 5.89 lb/sq.ft. for the spar caps. The upper and lower surface panels weigh 1.63 lb/sq.ft. and 1.37 lb/sq.ft., respectively, with the rib caps and webs, spar webs, and non-optimum factors having individual weights less than 0.5 lb/sq.ft. for all spar spacings.

SPANWISE STIFFENED WING ARRANGEMENT - TASK I

An Initial Screening and a Detailed Concept Analysis were conducted on the spanwise stiffened wing structural arrangement. The four panel concepts evaluated during the Initial Screening are presented in Figure 12-18. Also included on this figure is a typical wing box segment depicting the components included in the detailed concepts analysis. These components are: the upper and lower surface panels, spar caps and webs, rib caps and webs, and the appropriate non-optimum factors.

Minimum gages and other fabrication limits are summarized in Figure 12-19 for the spanwise panel concepts. A more detail description of this data is contained in the materials and producibility section, Section 7.

The load-temperature environment for the spanwise wing arrangement is based on the internal loads resulting from a NASTRAN redundant analysis solution. This solution utilized the 2-D finite element model with flexibilities representative of a typical spanwise stiffened wing. A description of the model and the input data is contained in Section 9, entitled Structural Analysis Models. Since the resulting internal loads are typical values, the point design environment was invariant and was used to analyze all the spanwise arrangements.

The point design environment was defined from a comprehensive list of flight conditions, see Section 11. The load-temperature environment for the most critical Task I flight condition is presented in Table 12-18. This condition is the Mach 1.25 symmetric flight condition at stall speed.

Spanwise Initial Screening

The spanwise stiffened wing panels are uniaxially stiffened panels which are of two basic constructions: integral and non-integral stiffened sheet. A total of four panel concepts, two of each basic construction, were included in the initial screening analysis; these concepts were:

- Zee stiffened
- Hat stiffened
- Integral zee, and the
- Integrally stiffened

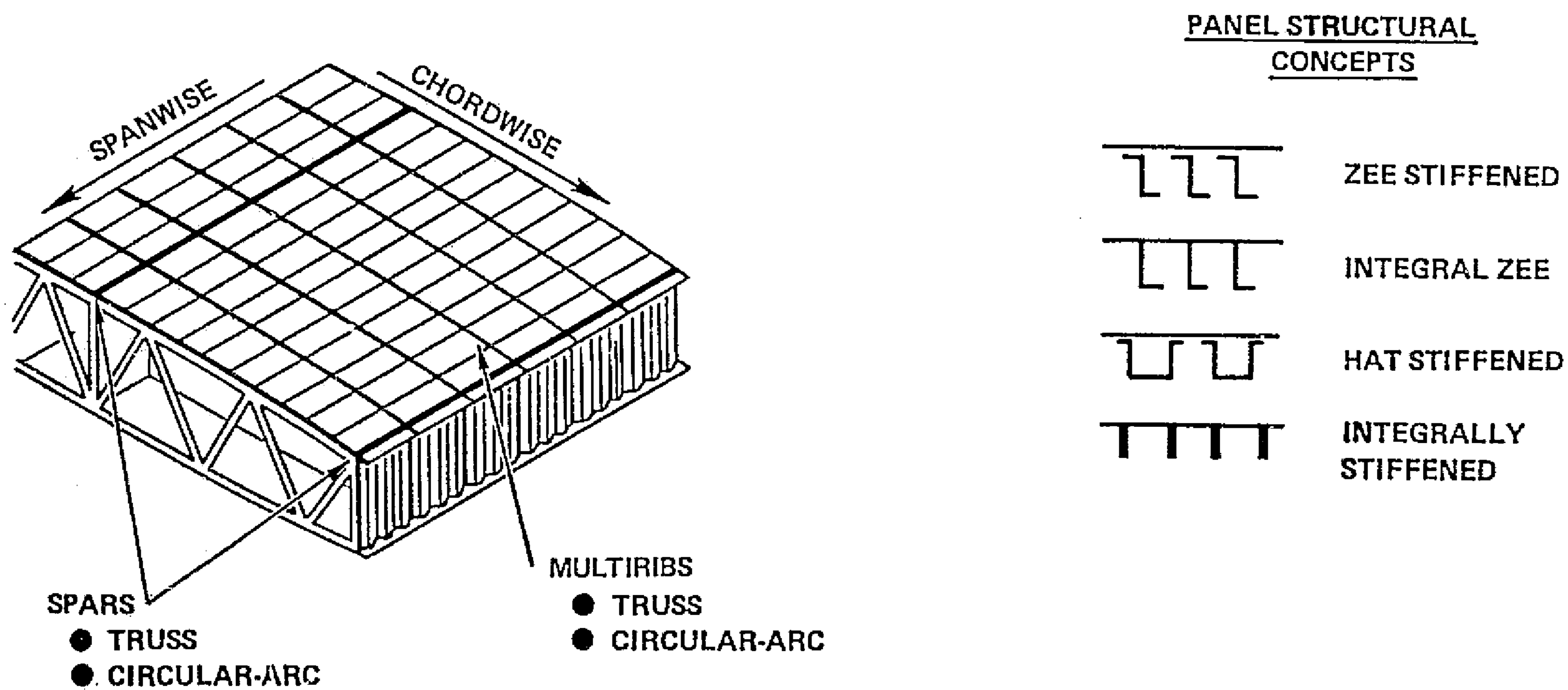
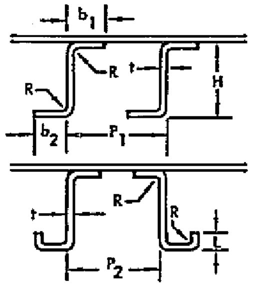
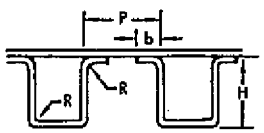


Figure 12-18. Spanwise Stiffened Wing Structural Arrangements

CONCEPT	FACE MATERIAL	INNER STIFF MATERIAL	SKIN GAUGES	SHEET SIZE	OUTER PANEL SIZE	FORMING LIMITS FOR STIFFENERS Ti 6Al-4V HOT FORMED BETA C COLD FORMED AND AGED: R AND t	P_1 AND t	b_1 AND b_2	H	L	FOREIGN OBJECT DAMAGE LIMITS *	METHOD OF ATTACH INNER TO OUTER SURFACE	CONTOUR LIMITS	COMMENT
	Ti 6 Al-4V ANNEALED OR BETA ALLOY STA	Ti 6 Al-4V ANNEALED OR BETA ALLOY STA	MIN .010 MAX .125 MIN .010 MAX .125	36 x 144 60 x 200 24 x 144 60 x 200	15' x 35' WELD FROM SMALLER SIZE SHEETS	MIN t .015 MIN BEND RAD. 3t	P_1 .75 MIN WITH b_2 .25 P_2 .65 MIN SET TO ALLOW MECH ATTACH OF RIB CLIP	b_1 .40 MIN SET TO GIVE BOND AND/OR BRAZE AREA b_2 .25 MIN	.50 IN. MIN PRACTICAL DIMENSION CONSISTENT WITH b DIMENSION	.15 MIN	t_o, OUTER SKIN WING UPPER SURFACE .015 WING LOWER SURFACE .020	WELD BOND CAPILLARY ACTION OR WELD BRAZE	INNER STIFFENER STRETCHED FORMED	SKINS WILL HAVE K_T INDUCED BY WELD QUALITY INDEX K_Q = 4 ASSUMED
	AS ABOVE		AS ABOVE	AS ABOVE	AS ABOVE		MIN .80 FOR INDIVIDUAL HATS .60 FOR CONTINUOUS HATS AS ABOVE	MIN .40 AS ABOVE	MIN .5 AS ABOVE	N/A	AS ABOVE			

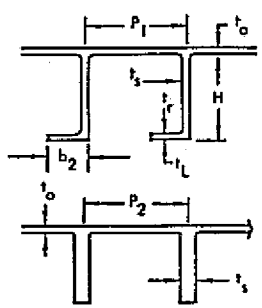
CONCEPT	MATERIAL	EXTRUSION BEFORE MACHINING	LIMITS AFTER MACHINING AND/OR CHEMICAL MILL			CONTOUR LIMITS	COMMENTS
	Ti 6 Al-4V ANNEALED OR BETA ALLOY MACHINED AND CHEM-MILLED FROM EXTRUSION	22 IN. MAXIMUM WIDTH .37 IN. MINIMUM THICKNESS PROJECTED .19 MIN THICKNESS $\frac{P_1}{H} > 1$ $\frac{P_2}{H} > 1$	AFTER MACHINING .040 MIN t_s AND t_L	AFTER CHEM MILL .020 MIN (STIFFENER AND SURFACE) t_s AND t_L	P_1 .75 MIN WITH MIN b_2 .25 P_2 .60 MIN SET TO ALLOW MECH ATTACH OF RIB CLIP NOTE: WELD SECTIONS TO GIVE FULL WIDTH PANEL PLUS STRESS RELIEVE	AS ABOVE HOT STRETCH FORM (1450°F) PLUS SHOT PEEN	CHEM MILL WILL EFFECT FATIGUE ALLOWABLE UNLESS SHOT PEENED K_Q = 4 SHOT PEENED K_Q = 5 WITHOUT SHOT PEEN

Figure 12-19. Fabrication Limits - Spanwise Stiffened Surface Panels

ORIGINAL PAGE IS OF POOR QUALITY

TABLE 12-18. WING POINT DESIGN ENVIRONMENT, SPANWISE ARRANGEMENT - TASK I

MASS - 313 X 10^3 Kg

CONDITION (31); MACH NO. = 1.25; n_z = 2.5

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION 40636 UPPER SURFACE	40636 LOWER SURFACE	41348 UPPER SURFACE	41348 LOWER SURFACE	40322 UPPER SURFACE	40322 LOWER SURFACE	40238 UPPER SURFACE	40238 LOWER SURFACE	41036 UPPER SURFACE	41036 LOWER SURFACE	41316 UPPER SURFACE	41316 LOWER SURFACE
AIR LOADS	N_x	kN/m	-91	91	-180	180	2	-2	54	-54	-79	79	28	-28
	N_y	kN/m	-2874	2874	-1648	1648	-208	208	-2975	2975	-1663	1663	-3143	3143
	N_{xy}	kN/m	731	731	482	482	51	51	445	445	565	565	752	752
THERMAL STRAIN	ϵ_x	m/m	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	m/m	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	m/m	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	kPa	-20.89	-8.27	-8.76	0.76	-34.33	-1.79	-8.76	-1.79	-34.96	6.62	-10.13	0.41
	FUEL	kPa	-40.88	-61.64	0	0	0	0	-39.09	-55.36	0	0	-47.30	-62.05
	NET	kPa	-61.77	-69.91	-8.76	0.76	-34.33	-1.79	-47.85	-57.15	-34.96	6.62	-57.43	-62.46
TEMPERATURE	T_{AV}	°K	334	334	354	359	342	340	333	333	350	355	344	345
	ΔT	°K	-74	-77	-62	-57	-79	-88	-72	-75	-60	-59	-63	-70

WEIGHT - [illegible] X 10^3 LB.

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION 40636 UPPER SURFACE	40636 LOWER SURFACE	41348 UPPER SURFACE	41348 LOWER SURFACE	40322 UPPER SURFACE	40322 LOWER SURFACE	40238 UPPER SURFACE	40238 LOWER SURFACE	41036 UPPER SURFACE	41036 LOWER SURFACE	41316 UPPER SURFACE	41316 LOWER SURFACE
AIR LOADS	N_x	LB/IN	-518	518	-1.028	1.0[illegible]	11	-11	306	-306	-450	450	162	-162
	N_y	LB/IN	-16.409	16.409	-9,412	9,412	-1.1[illegible]5	1,1[illegible]5	-16.986	16.986	-9.499	9,499	-17,948	17,948
	N_{xy}	LB/IN	4,173	4,173	2,750	[illegible],750	290	290	2,541	2,541	3,227	3,227	4,292	4,292
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	PSI	-1.27	-0.[illegible]6	-5.[illegible]7	0.96	-1.4[illegible]	0.06	-3.03	-1.20	-1.[illegible]7	0.11	4.98	-0.26
	FUEL	PSI	-5.67	-8.03	0	0	-[illegible].[illegible]0	-9.00	-5.93	-[illegible].94	0	0	0	0
	NET	PSI	-6.9	-8.29	-5.07	0.96	-[illegible].33	-[illegible].94	-8.96	-10.14	-1.27	0.11	4.98	-0.26
TEMPERATURE	T_{AV}	°F	141	142	177	18[illegible]	156	152	139	139	171	180	160	162
	ΔT	°F	-133	-139	-111	-10[illegible]	-143	-159	-130	-136	-108	-106	-114	-126

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

A weight-strength analysis was conducted on these panel concepts at the three point design regions: 40322, 40536, and 41348. See Figure 12-3 for the location of these regions on the wing platform.

The critical design condition and the corresponding load-temperature environment for this condition are presented in Table 12-18 with the pertinent regions used for the initial screening analysis noted.

The weight-strength analysis was conducted on both upper and lower surface panels for variable rib spacings of 20-, 30-, and 40-inches and a constant spar spacing of 60-inches. This analysis was conducted using the methods as previously described and the results are presented in the following text.

Hat-Stiffened Panels - The results of the panel analysis are summarized in Table 12-19 which presents the panel cross-sectional dimensions and mass data for each of the rib spacings investigated. In addition to the above data, a sketch of the panel cross section is presented in the footnotes.

A minimum gage constraint (0.020-inches) was active for the lower surface panels at point design region 40322. The skin thickness ranged from 0.020-to 0.109-inches, while the stiffener thickness varied from 0.019-to 0.100-inches. Unit weight varied from 1.20 lb/sq.ft. to 2.50 lb/sq.ft. on point design region 40322 and from 3.40 lb/sq.ft. to 6.3 lb/sq.ft. for regions 40536 and 41348.

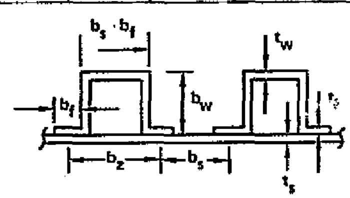
Zee Stiffened Panels - The results of the analysis conducted on this concept are summarized in Table 12-20 with a sketch showing the cross-sectional dimensions included in the footnotes.

As with the hat-stiffened concept, minimum skin gages (0.020-inches thick) are indicated for the lower surface panels at region 40322. For both panels at this region, the skin thickness ranged from the minimum gage value of 0.020-inches to 0.055-inches. The corresponding thickness range for region 40536 and 41348 was 0.060- to 0.114-inches.

The panels at region 40322 were the lightest weight designs ranging in unit weight from 1.14 lb/sq.ft. to 3.15 lb/sq.ft. Conversely, the heaviest panel designs occurred at region 40536 and varied from 5.28 lb/sq.ft. to 6.50 lb/sq.ft. The wing tip point design region 41348 indicated intermediate unit weight values ranging from 3.45 lb/sq.ft. to 5.06 lb/sq.ft.

TABLE 12-19. PANEL GEOMETRY AND WEIGHT FOR HAT SECTION STIFFENED CONCEPT – TASK I SPANWISE WING ARRANGEMENT

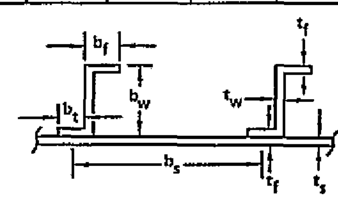
POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.0660	0.0890	0.1110	0.0520	0.0520	0.0530	0.2260	0.2390	0.2590	0.2770	0.2640	0.2670	0.1480	0.1710	0.2010	0.1600	0.1610	0.1610
	(in.)	0.0260	0.0350	0.0440	0.0200	0.0200	0.0210	0.0890	0.0940	0.1020	0.1090	0.1040	0.1050	0.0580	0.0672	0.0792	0.0630	0.0630	0.0640
$b_s = b_w = b_z$	(cm)	2.2800	3.2400	4.1700	2.0100	2.4600	2.8800	4.4600	5.3500	6.3700	5.1100	5.6900	6.4700	3.4100	4.4800	5.6100	3.5700	4.3400	5.0200
	(in.)	0.8960	1.2750	1.6420	0.7910	0.9690	1.1330	1.7570	2.1070	2.5090	2.0120	2.2410	2.5490	1.3440	1.7630	2.2090	1.4050	1.7110	1.9790
$t_w = t_f$	(cm)	0.0610	0.0830	0.1030	0.0480	0.0480	0.0490	0.2090	0.2210	0.2390	0.2550	0.2440	0.2460	0.1360	0.1570	0.1860	0.1470	0.1480	0.1490
	(in.)	0.0240	0.0320	0.0400	0.0190	0.0190	0.0190	0.0820	0.0870	0.0940	0.1000	0.0960	0.0970	0.0540	0.0620	0.0730	0.0580	0.0580	0.0590
b_f	(cm)	0.6830	0.9730	1.2500	0.[illegible]020	0.7390	0.8640	1.3400	1.6000	1.9100	1.5200	1.7100	1.9400	1.0200	1.3400	1.6800	1.0700	1.3000	1.5100
	(in.)	0.2690	0.3830	0.4930	0.2370	0.2910	0.3400	0.5270	0.6320	0.7530	0.6000	0.6720	0.7650	0.4030	0.5290	0.6630	0.4210	0.5130	0.5940
$b_s - b_f$	(cm)	1.5900	2.2700	2.9200	1.4100	1.7200	2.0100	3.1200	3.7500	4.4600	3.5600	3.9800	4.5300	2.3900	3.1300	3.9300	2.5000	3.0400	3.5[illegible]00
	(in.)	0.6270	0.8930	1.1500	0.5540	0.6780	0.7930	1.2300	1.4750	1.7570	1.4010	1.5690	1.7840	0.9410	1.2340	1.5470	0.9830	1.1980	1.2860
MASS DATA:																			
$\overline{t}$	(cm)	0.1669	0.2253	0.2804	0.1302	0.1302	0.1335	0.5716	0.6038	0.6544	0.6981	0.6665	0.6749	0.3735	0.4306	0.5074	0.4025	0.4056	0.4072
	(in.)	0.0657	0.0887	0.1104	0.0512	0.0512	0.0526	0.2250	0.2377	0.2577	0.2748	0.2624	0.2657	0.1470	0.1695	0.1998	0.1584	0.1597	0.1603
w	(kg/m^2)	7.3900	9.9800	12.4200	5.7700	5.7700	5.9100	25.3200	26.7400	28.9900	30.9000	29.5200	29.8900	16.5400	19.0700	22.4700	17.8200	17.9600	18.0400
	(lb/ft^2)	1.5140	2.0440	2.5430	1.1810	1.1810	1.2110	5.1850	5.4760	5.9370	6.3300	6.0460	6.1220	3.3880	3.9000	4.6020	3.6510	3.6750	3.6940
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



HAT SECTION STIFFENED

TABLE 12-20. PANEL GEOMETRY AND WEIGHT FOR ZEE STIFFENED PANEL CONCEPT

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.0800	0.1090	0.1400	0.0510	0.0510	0.0520	0.2350	0.2560	0.2900	0.2590	0.2640	0.2670	0.1540	0.1920	0.2260	0.1620	0.1630	0.1630
	(in.)	0.0310	0.0430	0.0550	0.0200	0.0200	0.0210	0.0930	0.1010	0.1140	0.1020	0.1040	0.1050	0.0600	0.0750	0.0890	0.0640	0.0640	0.0640
b_s	(cm)	2.8800	4.1300	5.4100	2.3000	2.8400	3.3000	5.2500	6.4000	7.7800	5.5800	6.5300	7.4600	4.0200	5.4800	6.8600	4.1400	5.0500	5.8200
	(in.)	1.1350	1.6270	2.1290	0.9070	1.1180	1.2990	2.0670	2.5190	3.0620	2.1960	2.5710	2.9360	1.5810	2.1560	2.6990	1.6290	1.9870	2.2900
$t_w = t_f$	(cm)	0.0840	0.1160	0.1490	0.0540	0.0550	0.0550	0.2490	0.2720	0.3070	0.2740	0.2790	0.2820	0.1630	0.2030	0.2390	0.1720	0.1720	0.1720
	(in.)	0.0330	0.0460	0.0580	0.0210	0.0220	0.0220	0.0980	0.1070	0.1210	0.1080	0.1100	0.1110	0.0640	0.0800	0.0940	0.0680	0.0680	0.0680
b_w	(cm)	2.5100	3.6000	4.7000	2.0000	2.4700	2.8700	4.5700	5.5700	6.7700	4.8500	5.6800	6.4900	3.4900	4.7600	5.9600	3.6000	4.3900	5.0600
	(in.)	0.9880	1.4160	1.8520	0.7890	0.9730	1.1300	1.7980	2.1920	2.6640	1.9100	2.2370	2.5540	1.3750	1.8760	2.3480	1.4170	1.7290	1.9920
b_f	(cm)	0.7500	1.0800	1.4100	0.6000	0.7400	0.8600	1.3700	1.6700	2.0300	1.4600	1.7000	1.9500	1.0500	1.4300	1.7900	1.0800	1.3200	1.5200
	(in.)	0.2960	0.4250	0.5560	0.2370	0.2920	0.3390	0.5390	0.6570	0.7990	0.5730	0.6710	0.7660	0.4130	0.5630	0.7050	0.4250	0.5190	0.5980
MASS DATA:																			
$\bar{t}$	(cm)	0.1972	0.2703	0.3468	0.1258	0.1275	0.1292	0.5823	0.6319	0.7166	0.6388	0.6526	0.6596	0.3604	0.4743	0.5576	0.4006	0.4029	0.4012
	(in.)	0.0776	0.1064	0.1365	0.0495	0.0502	0.0509	0.2292	0.2488	0.2821	0.2515	0.2569	0.2597	0.1498	0.1868	0.2195	0.1577	0.1586	0.1580
w	(kg/m^2)	8.7300	11.9700	15.3600	5.5700	5.6500	5.7200	25.7900	27.9900	31.7400	28.2900	28.9000	29.2100	16.8500	21.0100	24.7000	17.7400	17.8400	17.7700
	(lb/ft^2)	1.7890	2.4520	3.1460	1.1410	1.1570	1.1720	5.2820	5.7320	6.5000	5.7950	5.920	5.9830	3.4510	4.3030	5.0580	3.6340	3.6540	3.6390
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



ZEE STIFFENED

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Integral Zee Panels - The results of the panel sizing are shown in Table 12-21. With reference to this table, skin thicknesses ranged from the minimum gage value of 0.020-inches to 0.122-inches for the three point design regions.

The heaviest-weight panels occurred at point design region 40536, which varied from 5.13 lb/sq.ft. for the upper surface panel to 6.47 lb/sq.ft. for the maximum weight lower panel. The least-weight panels occurred at region 40322 and ranged from 1.06 lb/sq.ft. to 2.84 lb/sq.ft. The intermediate weight panels, region 41348, ranged from 3.22 lb/sq.ft. to 4.69 lb/sq.ft.

Integrally Stiffened Panel - The results of the panel analysis, which are shown in Table 12-22, indicate this concept is the most inefficient spanwise design from a weight/strength standpoint. With reference to this table, minimum gage skins are noted on the lower surface panels at point design region 40322, while the thickest skin gages occur on the surface panels at region 40536 where the corresponding stiffener thicknesses range from 0.046- to 0.31-inches.

The forward wing (region 40322) lower surface panels weigh approximately 1.20 lb/sq.ft., while the heaviest panels, approximately 6.50 lb/sq.ft. occur at point design region 40536. The panels at region 41348 range from approximately 4.0 lb/sq.ft. to 6.0 lb/sq.ft.

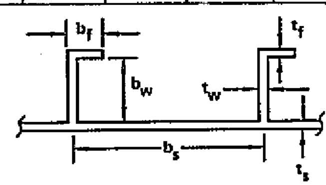
Spanwise Surface Panel Results - Comparison curves of the surface panel weights for the spanwise concepts are presented in Figure 12-20. These unit weights, sum of the upper and lower surface panel weights, are displayed as a function of rib spacing at each of the point design regions.

With reference to the forward wing box region 40322, the hat-stiffened concept is the least-weight panel concept at all rib spacing investigated, e.g., approximately 2.7 lb/sq.ft. at 20-inch rib spacing. Conversely, the integral stiffened concept is the heaviest design with a unit weight of 3.4 lb/sq.ft. for the 20-inch rib spacing. For the remaining concepts, the integral-zee and zee-stiffened concepts are ranked, with respect to weights, as the second and third best concepts, respectively.

The panels at point design region 40536 exhibit the same weight characteristics as those at region 40322, i.e., least-and heaviest-weight designs are the hat-stiffened and integral-stiffened panel concepts, respectively. The exception being the ranking of the concepts for the 20-inch rib spacing designs, for this

TABLE 12-21. PANEL GEOMETRY AND WEIGHT FOR THE INTEGRAL ZEE PANEL CONCEPT - TASK I
SPANWISE WING ARRANGEMENT

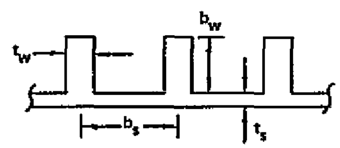
POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.0780	0.1080	0.1360	0.0510	0.0510	0.0600	0.2460	0.2620	0.2890	0.3070	0.3100	0.3020	0.1540	0.1900	0.2160	0.1820	0.1780	0.1780
	(in.)	0.0310	0.0420	0.0540	0.0200	0.0200	0.0240	0.0970	0.1030	0.1140	0.1210	0.1220	0.1190	0.0610	0.0750	0.0880	0.0720	0.0700	0.0700
b_s	(cm)	2.4900	3.5800	4.6300	2.0100	2.4600	3.0800	4.7400	5.6700	6.7700	5.4900	6.3500	6.9600	3.5200	4.7500	5.9600	3.9100	4.6000	5.3000
	(in.)	0.9800	1.4080	1.8230	0.7900	0.9680	1.2140	1.8650	2.2330	2.6650	2.1630	2.5000	2.7400	1.3840	1.8700	2.3480	1.5390	1.8110	2.0880
$t_w = t_f$	(cm)	0.0780	0.1080	0.1360	0.5100	0.0510	0.0600	0.2460	0.2620	0.2890	0.3070	0.3100	0.3020	0.1540	0.1900	0.2160	0.1820	0.1780	0.1780
	(in.)	0.0309	0.0424	0.0535	0.0200	0.0200	0.0237	0.0968	0.1030	0.1140	0.1210	0.1220	0.1190	0.0607	0.0749	0.0885	0.0718	0.0703	0.0700
b_w	(cm)	2.4900	3.5800	4.6300	2.0100	2.4600	3.0800	2.4900	3.5800	4.6300	2.0100	2.4600	3.0800	3.5200	4.7500	5.9600	3.9100	4.6000	5.3000
	(in.)	0.9800	1.4800	1.8230	0.7900	0.9680	1.2140	1.8650	2.2380	2.6650	2.1630	2.5000	2.7400	1.3840	1.8700	2.3490	1.5390	1.8110	2.0880
b_f	(cm)	0.7500	1.0700	1.3900	0.6000	0.7400	0.9200	1.4200	1.7100	2.0300	1.6500	1.9000	2.0800	1.0500	1.4200	1.7900	1.1700	1.3800	1.5900
	(in.)	0.2940	0.4220	0.5480	0.2370	0.2900	0.3640	0.5600	0.6720	0.8000	0.6500	0.7500	0.8210	0.4150	0.5610	0.7040	0.4610	0.5430	0.6260
MASS DATA:																			
$\bar{t}$	(cm)	0.1604	0.2481	0.3128	0.1173	0.1173	0.1383	0.5652	0.6005	0.6640	0.7088	0.7131	0.6950	0.3544	0.4377	0.5173	0.4197	0.4105	0.4090
	(in.)	0.0710	0.0977	0.1231	0.0462	0.0462	0.0545	0.2225	0.2364	0.2614	0.2791	0.2808	0.2736	0.1395	0.1723	0.2036	0.1652	0.1616	0.1610
W	(kg/m^2)	7.9900	10.9900	13.8500	5.1900	5.1900	6.1300	25.0300	26.5900	29.4000	31.3900	31.5800	30.7800	15.7000	19.3800	22.9100	18.5900	18.1800	18.1100
	(lb/ft^2)	1.6370	2.2510	2.8370	1.0640	1.0640	1.2550	5.1270	5.4470	6.0230	6.4300	6.4690	6.3040	3.2150	3.9700	4.6920	3.8[illegible]0	3.7230	3.7100
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



INTEGRAL ZEE

TABLE 12-22. PANEL GEOMETRY AND WEIGHT FOR THE INTEGRALLY STIFFENED PANEL CONCEPT - TASK I SPANWISE WING ARRANGEMENT

POINT DESIGN REGIONS		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.1000	0.1380	0.1760	0.0510	0.0510	0.0650	0.2560	0.3000	0.3530	0.2840	0.2900	0.3000	0.1840	0.2330	[illegible]	0.1800	0.1780	0.1760
	(in.)	0.0390	0.0540	0.0700	0.0200	0.0200	0.0260	0.1010	0.1180	0.1390	0.1120	0.1140	0.1180	0.0730	0.0920	[illegible]	0.0710	0.0700	0.0700
b_s	(cm)	4.2300	6.1000	7.9600	3.0500	3.7300	4.8400	6.9200	8.9900	11.2600	7.3900	8.8500	10.3700	5.7500	7.9200	9.9300	5.6900	5.7900	7.9600
	(in.)	1.6660	2.4010	3.1340	1.2000	1.4700	1.9050	2.7240	3.540	4.4320	2.9110	3.4850	4.0810	2.2640	3.1190	3.9210	2.2400	2.7290	3.1340
$t_W = t_f$	(cm)	0.2240	0.3100	0.3960	0.1160	0.1160	0.1470	0.5740	0.6760	0.7950	0.6380	0.6550	0.6730	0.4140	0.5230	0.6220	0.4060	0.4010	0.3960
	(in.)	0.0883	0.1220	0.1560	0.0459	0.0459	0.0578	0.2260	0.2660	0.3130	0.2510	0.2580	0.2650	0.1630	0.2060	0.2450	0.1600	0.1580	0.1560
b_W	(cm)	2.7500	3.9600	5.1700	1.9800	2.4300	3.1400	4.5000	5.8400	7.3200	4.8000	5.7500	6.7400	3.7400	5.1500	6.4700	3.7000	4.5000	5.1700
	(in.)	1.0830	1.5610	2.0370	0.7800	0.9560	1.2380	1.7700	2.3010	2.8810	1.8920	2.2650	2.6530	1.4720	2.0270	2.5480	1.4560	1.7740	2.0370
MASS DATA:																			
$\overline{t}$	(cm)	0.2460	0.3400	0.4340	0.1270	0.1270	0.1600	0.6680	0.7380	0.8690	0.6980	0.7160	0.7370	0.4530	0.5740	0.6800	0.4440	0.4390	0.4340
	(in.)	0.0967	0.1338	0.1710	0.0502	0.0502	0.0632	0.2475	0.2908	0.3421	0.2746	0.2819	0.2900	0.1785	0.2258	0.2677	0.1740	0.1729	0.1710
w	(kg/m^2)	10.8700	18.7400	19.2400	5.6400	5.6400	7.1100	27.8400	32.7100	38.4800	30.8900	31.7100	32.6200	20.0800	25.4100	30.1100	19.6600	19.4500	19.2400
	(lb/ft^2)	2.2270	3.0840	3.9400	1.1560	1.1560	1.4560	5.702	6.6990	7.8810	6.3270	6.4950	6.6820	4.1130	5.2040	6.1680	4.0270	3.9830	3.9410
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



INTEGRALLY STIFFENED

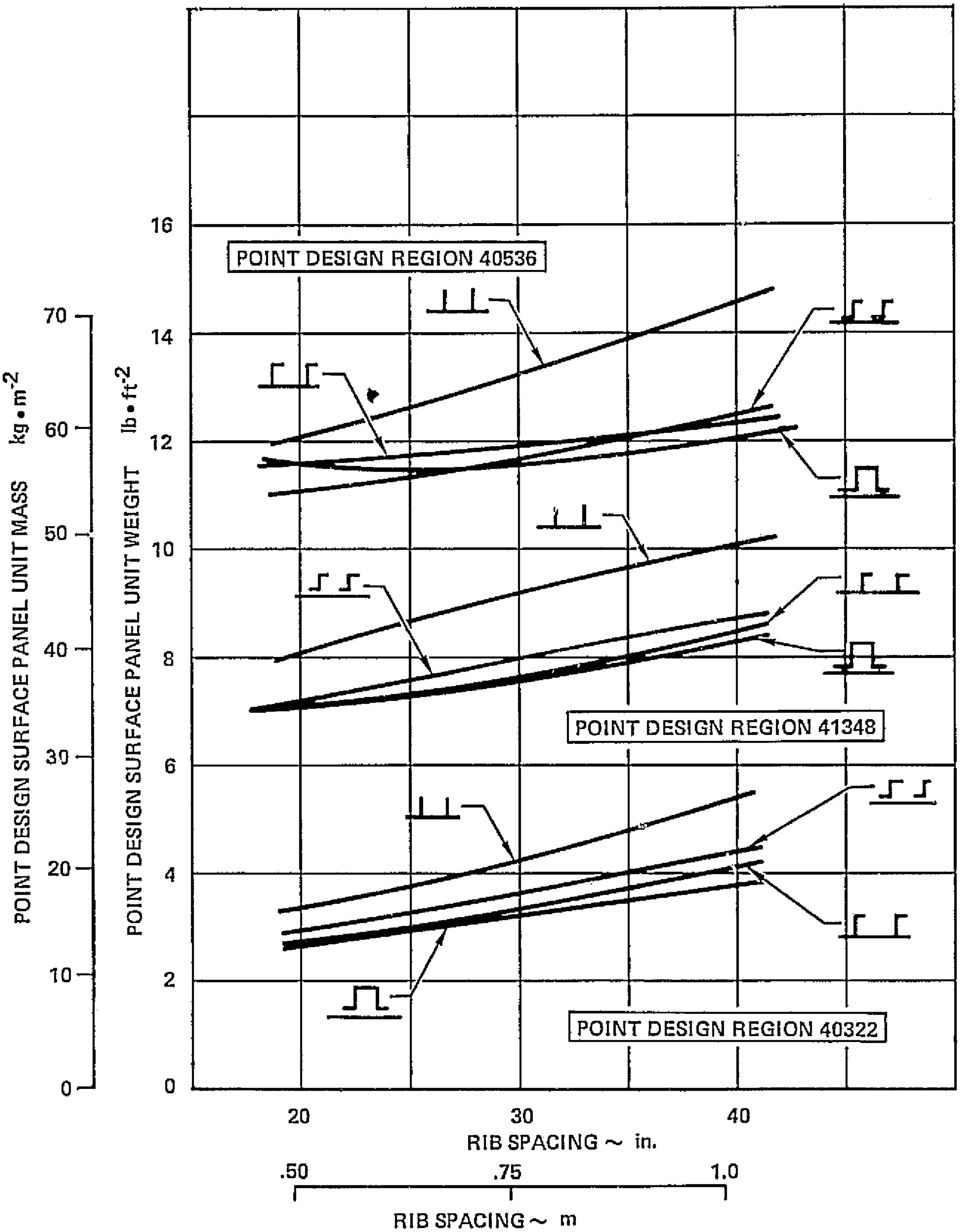


Figure 12-20. Weight Comparison of the Candidate Spanwise Panel Concepts

spacing the zee-stiffened concept supersedes the hat-stiffened concept as the least weight design. The unit weight values for the lightest (zee-stiffened) and heaviest (integrally-stiffened) concepts for the 20-inch rib spacing designs are approximately 11.0 lb/sq.ft. and 12.0 lb/sq.ft., respectively. In general, the zee-stiffened and integrally-zee concepts are ranked second and third with unit weights ranging from 11.0 lb/sq.ft. to 12.5 lb/sq.ft. for the various rib spacings.

The surface panel weight curves for the wing tip region 41348 are the center curves shown in Figure 12-20. With respect to these curves, the ranking of the panel concepts on a weight basis is: (1) least-weight hat-stiffened concept, (2) integral zee, (3) zee-stiffened, and (4) heaviest-weight integrally stiffened concept. This ranking holds for all rib spacings. For comparison purposes, the least-weight hat stiffened design and heaviest-weight integrally stiffened design have respective unit weights values of 7.0 lb/sq.ft. and 8.1 lb/sq.ft. for the 20-inch rib spacing design.

Spanwise Detailed Concept Analysis

The most promising panel concept surviving the spanwise initial screening analysis was the hat-stiffened concept. This panel concept was subjected to point design analysis at six wing regions, the three regions investigated during the initial screening plus three additional regions located in the wing aft box and wing tip. Figure 12-3 indicates the locations of the point design regions used for this analysis.

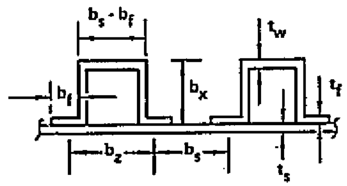
In addition to analyzing this concept at more locations, the weight-strength analysis was conducted in more depth and included determining unit box weights at each of the six point design regions, i.e., weights of surface panels, substructure, and non-optimum factors.

The surface panel load-temperature environment for the most critical flight condition at each point design region was as previously shown in Table . 2-18. In addition, the panel fabrication limits defined in Figure 12-19 are also applicable for this analysis.

Panel Analysis - The results of the weight-strength analysis at the three new point design regions are presented in Table 12-23. This table summarizes the panel dimensions and weights for each of rib spacings studied; 20-, 30-, and 40-inches. A constant spar spacing of 60-inches was maintained.

TABLE 12-23. PANEL GEOMETRY AND WEIGHT FOR THE HAT SECTION STIFFENED PANEL CONCEPT - TASK I SPANWISE WING ARRANGEMENT - DETAIL CONCEPT ANALYSIS

POINT DESIGN REGIONS		40236						41036						41316					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.2090	0.2290	0.2650	0.2260	0.2250	0.2300	0.1600	0.1780	0.1960	0.1850	0.1810	0.1800	0.2490	0.2540	0.2730	0.2760	0.2770	0.2730
	(in.)	0.0820	0.0900	0.1040	0.0890	0.0890	0.0910	0.0630	0.0700	0.0770	0.0730	0.0710	0.0710	0.0980	0.1000	0.1080	0.1090	0.1090	0.1080
$b_s = b_w = b_z$	(cm)	4.2400	5.2200	6.4400	4.4600	5.1700	6.0000	3.5700	4.5800	5.5400	3.9100	4.6000	5.3100	4.7400	5.5500	6.5400	5.0800	5.8600	6.5400
	(in.)	1.6680	2.0540	2.5360	1.7570	2.0340	2.3640	1.4050	1.8030	2.1800	1.5400	1.8130	2.0900	1.8670	2.1860	2.5750	2.0020	2.3090	2.5750
$t_w = t_f$	(cm)	0.1930	0.2120	0.2440	0.2090	0.2080	0.2150	0.1470	0.1650	0.1810	0.1700	0.1670	0.1660	0.2300	0.2340	0.2520	0.2550	0.2560	0.2520
	(in.)	0.0760	0.0830	0.0960	0.0870	0.0820	0.0840	0.0580	0.0650	0.0710	0.0670	0.0660	0.0650	0.0900	0.0920	0.0990	0.1000	0.1010	0.0990
b_f	(cm)	1.2700	1.5600	1.9300	1.3400	1.5500	1.8000	1.0700	1.3700	1.6600	1.1700	1.3800	1.5900	1.4200	1.6700	1.9600	1.5200	1.7600	1.9600
	(in.)	0.5000	0.6160	0.7610	0.5270	0.6100	0.7090	0.4210	0.5410	0.6540	0.46[illegible]	0.5440	0.6270	0.5610	0.6560	0.7730	0.6000	0.6930	0.7730
$b_s - b_f$	(cm)	2.9700	3.6500	4.5100	3.1200	3.6200	4.2000	2.5000	3.2000	3.8800	2.7400	3.2200	3.7200	3.3200	3.8900	4.5800	3.5600	4.1100	4.5800
	(in.)	1.1680	1.4380	1.7750	1.2300	1.4240	1.6550	0.9830	1.2620	1.5260	1.0780	1.2690	1.4630	1.3070	1.5300	1.8030	1.4010	1.6170	1.8030
MASS DATA:																			
$\bar{t}$	(cm)	0.5277	0.5786	0.7389	0.5716	0.5688	0.5808	0.4025	0.4506	0.4940	0.4659	0.4556	0.4539	0.6273	0.6409	0.6889	0.6981	0.6988	0.6889
	(in.)	0.2078	0.2278	0.2941	0.2250	0.2239	0.2287	0.1584	0.1774	0.1945	0.1834	0.1794	0.1787	0.2470	0.2523	0.2712	0.2748	0.2751	0.2712
w	(kg/m^2)	23.3700	25.6200	33.0800	25.3200	25.1900	25.7200	17.8200	19.9600	21.8800	20.6300	20.1800	20.1100	27.7800	28.3900	30.5100	30.9000	30.9400	30.5100
	(lb/ft^2)	4.7900	5.2500	6.7800	5.1800	5.1600	5.2700	3.6500	4.0900	4.4800	4.2300	4.1300	4.1200	5.6900	5.8100	6.2500	6.3300	6.3400	6.2500
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



HAT SECTION STIFFENED

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Skin thicknesses varied from 0.063-inches to 0.109-inches with the stiffener thickness ranging from 0.058-inches to approximately 0.100-inches.

Region 41036 had the least-weight panels for the new point design regions, unit weights for this region varied from 3.65 lb/sq.ft. to 4.46 lb/sq.ft. for the upper surface panels. The heaviest-weight panels occurred at point design region 41316, inboard region of the wing tip, where the upper surface panels varied in weight from 5.69 lb/sq.ft. to 6.25 lb/sq.ft. And the tension design lower surface had an average unit weight of approximately 6.30 lb/sq.ft. The upper surface of region 40236, inboard region on the aft wing box, experienced the largest variation in unit weight, ranging from 4.79 lb/sq.ft. to 6.78 lb/sq.ft. This variation is attributed to the panels being predominately designed by the high compressive loads. The tension design lower surface panel at region 40236 experienced a slight variation in unit weight (0.11 lb/sq.ft.) with an average weight of approximately 5.20 lb/sq.ft.

The surface panel designs established for the hat-stiffened concept during the initial screening study are applicable for this analysis. The panel dimensions and unit weights for these regions (40322, 40526, and 41348) were previously presented in Table 12-19.

Substructure Analysis - For the spanwise stiffened wing arrangement the surface panels carry the wing spanwise bending loads with the rib caps supporting the chordwise loads. The chordwise loads resisted by the skin were conservatively neglected. Both truss and circular-arc webs were considered for the spar and rib web design. Since the panels are the main spanwise load carrying members, only light spar caps located at contour are required.

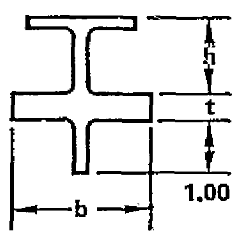
All substructure components were subjected to analysis at each point design region and the resulting weights are summarized in the detail wing weights reported later in the wing box results. For this section, only the results of the rib cap analysis (geometry and weight) are reported to illustrate the depth of analysis conducted on the substructure components. Table 12-24 contains this data for the upper and lower rib caps at each of the point design regions. A sketch of the rib cap design is included in the footnotes.

Spanwise Box Weights - A compilation of the component and total wing box weights for the spanwise arrangement at each point design region are shown in Table 12-25 and 12-26. These tables includes the weight of the surface panels, rib webs, spar

TABLE 12-24. RIB CAP GEOMETRY AND WEIGHT FOR THE SPANWISE WING ARRANGEMENT

POINT DESIGN REGION	SPACING		UPPER CAP DATA					LOWER CAP DATA				
	SPAR (IN.)	RIB (IN.)	h (IN.)	b (IN.)	$\bar{t}$ (IN.)	A ($IN.^2$)	$\bar{t}$ (IN.)	h (IN.)	b (IN.)	t (IN.)	A ($IN.^2$)	t (IN.)
40322	60	20	1.00	2.0	.060	.120	.0060	1.00	2.0	.060	.120	.0060
		30	1.38	2.0	.060	.120	.0040	1.07	2.0	.060	.120	.0040
		40	1.74	2.0	.060	.120	.0030	1.23	2.0	.060	.120	.0030
40536	60	20	1.70	2.0	.060	.120	.0060	2.11	2.0	.067	.134	.0067
		30	2.21	2.0	.072	.144	.0048	2.34	2.0	.105	.210	.0070
		40	2.61	2.0	.099	.198	.0050	2.65	2.0	.144	.288	.0072
41348	60	20	1.39	2.0	.104	.208	.0104	1.51	2.0	.156	.312	.0156
		30	1.87	2.0	.175	.350	.0117	1.81	2.0	.253	.506	.0169
		40	2.31	2.0	.261	.522	.0131	2.08	2.0	.379	.758	.0190
40236	60	20	1.80	2.0	.060	.120	.0060	1.90	2.0	.060	.120	.0060
		30	2.20	2.0	.060	.120	.0040	2.20	2.0	.060	.120	.0040
		40	2.70	2.0	.060	.120	.0030	2.50	2.0	.060	.120	.0030
41036	60	20	1.60	2.0	.060	.120	.0060	1.70	2.0	.060	.120	.0060
		30	2.00	2.0	.090	.180	.0060	2.00	2.0	.065	.130	.0043
		40	2.30	2.0	.131	.262	.0066	2.30	2.0	.091	.182	.0046
41316	60	20	2.00	2.0	.061	.122	.0061	2.20	2.0	.060	.120	.0060
		30	2.30	2.0	.092	.184	.0061	2.50	2.0	.061	.122	.0041
		40	2.70	2.0	.138	.276	.0069	2.70	2.0	.095	.190	.0048

CAP GEOMETRY



NOTES:

AREA (A) = b x t

$\bar{t}$ = A/RIB SPACING

AREA OF THE CLIP AND WEB ATTACHMENT INCLUDED IN NON-OPTIMUM WEIGHTS.

TABLE 12-25. DETAIL WING WEIGHTS FOR THE SPANWISE WING ARRANGEMENT

POINT DESIGN REGION	40322			41316			41348		
RIB SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	1.514	2.044	1.543	5.690	5.814	6.249	3.388	3.906	4.602
LOWER	1.181	1.181	1.211	6.330	6.339	6.249	3.651	3.679	3.694
Σ	(2.695)	(3.225)	(3.754)	(12.020)	(12.153)	(12.498)	(7.039)	(7.585)	(8.296)
RIB WEBS									
BULKHEAD	0.517	0.551	0.555	0.271	0.218	0.181	0.276	0.187	0.159
TRUSS	0.409	0.373	0.296	—	—	—	—	—	—
Σ	(0.926)	(0.924)	(0.851)	(0.271)	(0.218)	(0.181)	(0.276)	(0.187)	(0.159)
SPAR WEBS									
BULKHEAD	0.133	0.133	0.133	0.163	0.163	0.163	0.200	0.200	0.200
TRUSS	0.230	0.230	0.230	—	—	—	—	—	—
Σ	(0.363)	(0.363)	(0.363)	(0.163)	(0.163)	(0.163)	(0.200)	(0.200)	(0.200)
RIB CAPS									
UPPER	0.139	0.105	0.070	0.139	0.147	0.159	0.240	0.270	0.302
LOWER	0.139	0.105	0.070	0.139	0.101	0.108	0.360	0.390	0.583
Σ	(0.278)	(0.210)	(0.140)	(0.278)	(0.248)	(0.267)	(0.600)	(0.660)	(0.885)
SPAR CAPS									
UPPER	0.069	0.069	0.076	0.357	0.247	0.202	0.063	0.079	0.093
LOWER	0.069	0.069	0.069	0.395	0.268	0.202	0.073	0.075	0.074
Σ	(0.138)	(0.138)	(0.145)	(0.752)	(0.515)	(0.404)	(0.136)	(0.154)	(0.167)
NON-OPTIMUM									
FAST./CLIPS	0.180	0.170	0.160	0.200	0.190	0.180	0.200	0.190	0.180
WEB INTERS.	0.129	0.129	0.121	0.043	0.038	0.034	0.048	0.039	0.036
	(0.309)	(0.299)	(0.281)	(0.243)	(0.228)	(0.214)	(0.248)	(0.229)	(0.216)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	4.709	5.159	5.534	13.727	13.525	13.727	8.499	9.015	9.923

TABLE 12-26. DETAIL WING WEIGHTS FOR THE SPANWISE WING ARRANGEMENT

POINT DESIGN REGION	40236			40536			41036		
RIB SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	4.787	5.248	6.776	5.185	5.476	5.937	3.651	4.088	4.481
LOWER	5.185	5.159	5.268	6.330	6.046	6.122	4.226	4.133	4.118
Σ	(9.972)	(10.407)	(12.044)	(11.515)	(11.522)	(12.059)	(7.877)	(8.221)	(8.599)
RIB WEBS									
BULKHEAD	0.451	0.513	0.559	0.396	0.422	0.425	0.150	0.177	0.204
TRUSS	0.642	0.423	0.300	0.410	0.280	0.187	0.175	0.124	0.073
Σ	(0.093)	(0.936)	(0.859)	(0.806)	(0.702)	(0.612)	(0.325)	(0.301)	(0.277)
SPAR WEBS									
BULKHEAD	0.353	0.353	0.353	0.380	0.380	0.380	0.072	0.072	0.072
TRUSS	0.080	0.080	0.080	0.125	0.125	0.125	0.040	0.040	0.040
Σ	(0.433)	(0.433)	(0.433)	(0.505)	(0.505)	(0.505)	(0.112)	(0.112)	(0.112)
RIB CAPS									
UPPER	0.139	0.092	0.092	0.139	0.110	0.115	0.139	0.145	0.152
LOWER	0.139	0.092	0.069	0.155	0.161	0.160	0.139	0.100	0.104
Σ	(0.278)	(0.184)	(0.161)	(0.294)	(0.271)	(0.275)	(0.278)	(0.245)	(0.256)
SPAR CAPS									
UPPER	0.296	0.219	0.213	0.120	0.116	0.159	0.227	0.171	0.143
LOWER	0.319	0.215	0.166	0.169	0.154	0.196	0.260	0.173	0.132
Σ	(0.615)	(0.434)	(0.379)	(0.289)	(0.270)	(0.355)	(0.487)	(0.344)	(0.275)
NON-OPTIMUM									
FASTENERS/CUPS	0.200	0.190	0.180	0.200	0.190	0.190	0.200	0.190	0.180
WEB INTERS.	0.152	0.136	0.130	0.131	0.121	0.112	0.044	0.041	0.030
Σ	(0.352)	(0.326)	(0.310)	(0.331)	(0.311)	(0.292)	(0.244)	(0.231)	(0.210)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	12.743	12.720	14.186	13.740	13.581	14.098	9.323	9.454	9.729

webs, rib caps, spar caps, and associated non-optimum factors. For easier interpretations these results are displayed in graphic form in Figures 12-21 through 12-26.

The component weights and total box weight for the forward wing box point design region 40322 are presented in Figure 12-21. The upper surface panel displays a large positive slope characteristics of panels designed by high compression loads in combination with normal pressure, i.e., beam column effect. The panel weight ranged from 1.5 to 2.5 lb/sq.ft. for 20- and 40-inch rib spacing, respectively. The lower surface panel weights. which are tension designed, indicate only a slight variation with rib spacing and has a value of approximately 1.20 lb/sq.ft. The weights for all other components display only a slight variation with rib spacing with the largest weight attributed to the rib webs, approximately 0.90 lb/sq.ft. With reference to the total weight curve on this figure, the minimum-weight design is coincidental with the smallest rib spacing investigated with the total weight varying from approximately 4.7 lb/sq.ft. to 5.5 lb/sq.ft. for the 20- and 40-inch rib spacing designs, respectively.

With reference to Figure 12-22, a minimum-weight design of 12.72 lb/sq.ft. is indicated for region 40236 at a rib spacing of 30-inches. A slightly higher total box weight is noted for the 20-inch rib spacing design (12.74 lb/sq.ft.), while a much larger weight increase is shown for the 40-inch design, i.e., approximately 12-percent increase over the minimum-weight design. The weight curves for the surface panel design exhibit the same characteristic slopes as those indicated for region 40322. The upper surface panel weights vary from 4.8 lb/sq.ft. to 6.8 lb/sq.ft. for the 20-inch and 40-inch rib spacing designs, while the weights of the lower surface panels were almost invariant with respect to rib spacing at approximately 5.2 lb/sq.ft. All other components indicated negligible weight changes with respect to rib spacing with the largest weight component being the rib webs at approximately 1.0 lb/sq.ft.

The point design box weights for region 40536 are shown in Figure 12-23. This region, which is located at approximately mid-span on the wing aft box, has a minimum-weight design of 13.58 lb/sq.ft. (total box weight) for the 30-inch rib spacing design. The corresponding total box weights for the 20-inch and 40-inch designs are 13.74 lb/sq.ft. and 14.10 lb/sq.ft., respectively. As compared with the prior regions, a smaller weight increment is noted between the upper and

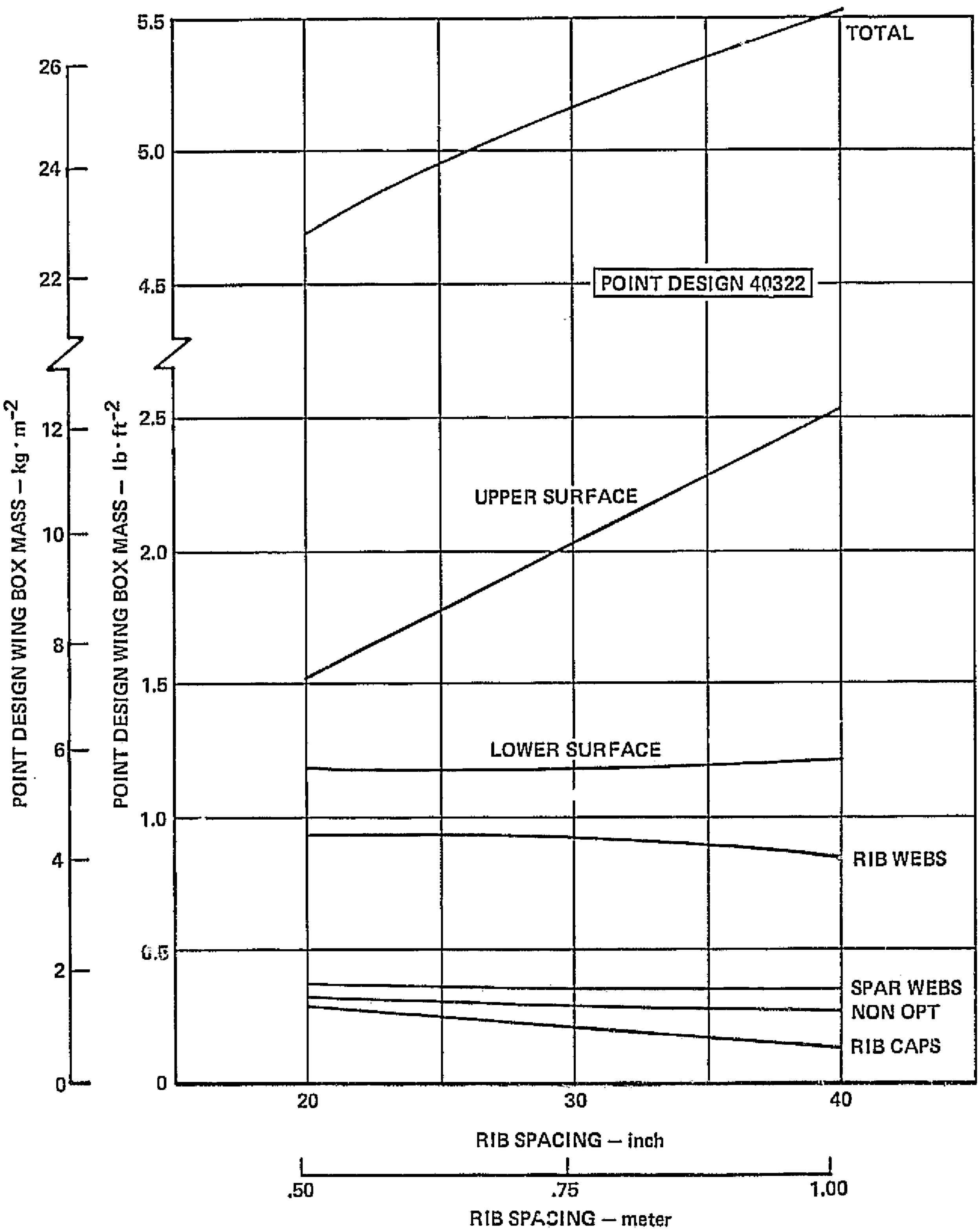


Figure 12-21. Optimum Rib Spacing for Spanwise Wing Arrangement at Point Design Region 40322

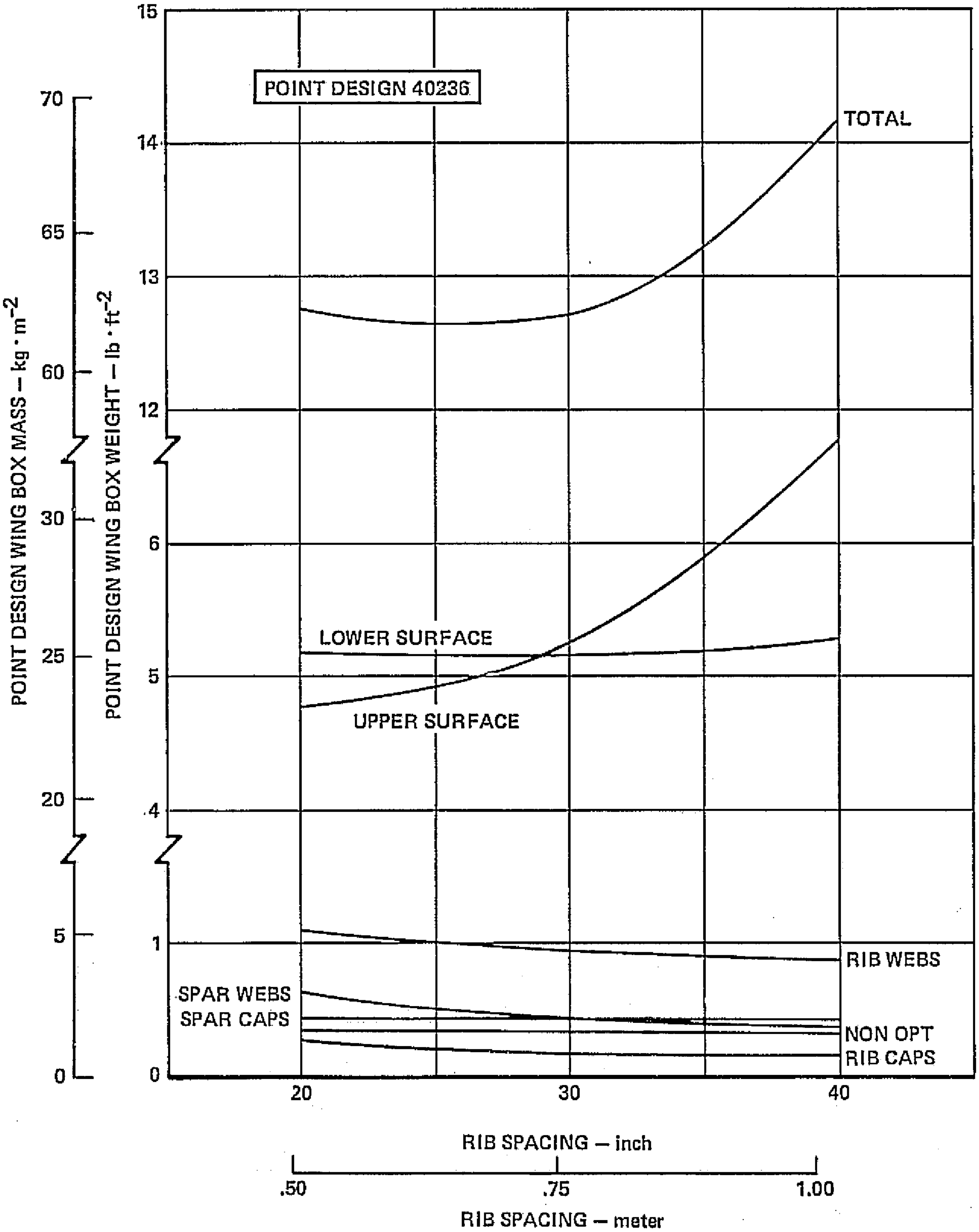


Figure 12-22. Optimum Rib Spacing for Spanwise Wing Arrangement at Point Design Region 40236

C . 2

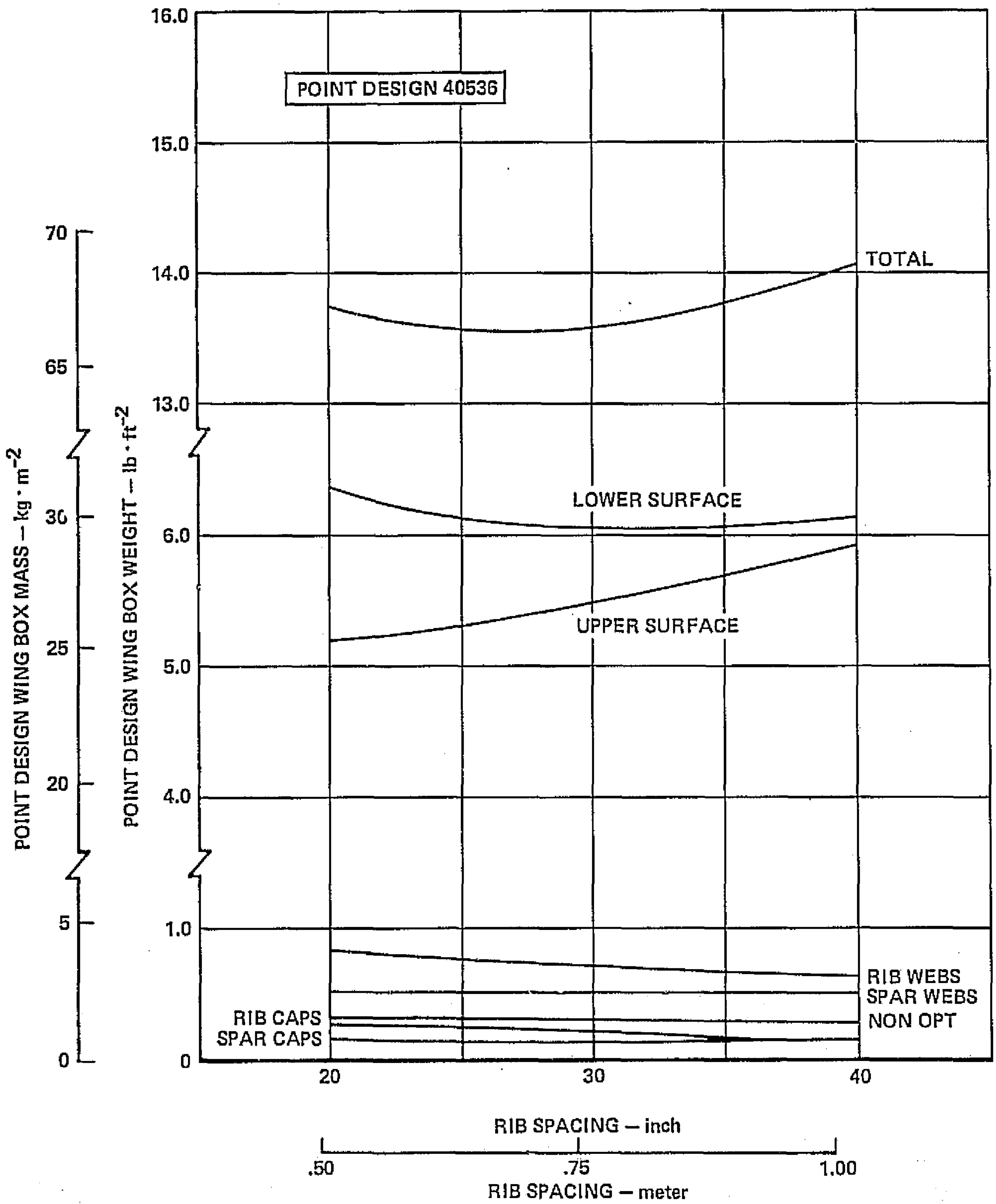


Figure 12-23. Optimum Rib Spacing for Spanwise Wing Arrangement at Point Design Region 40536

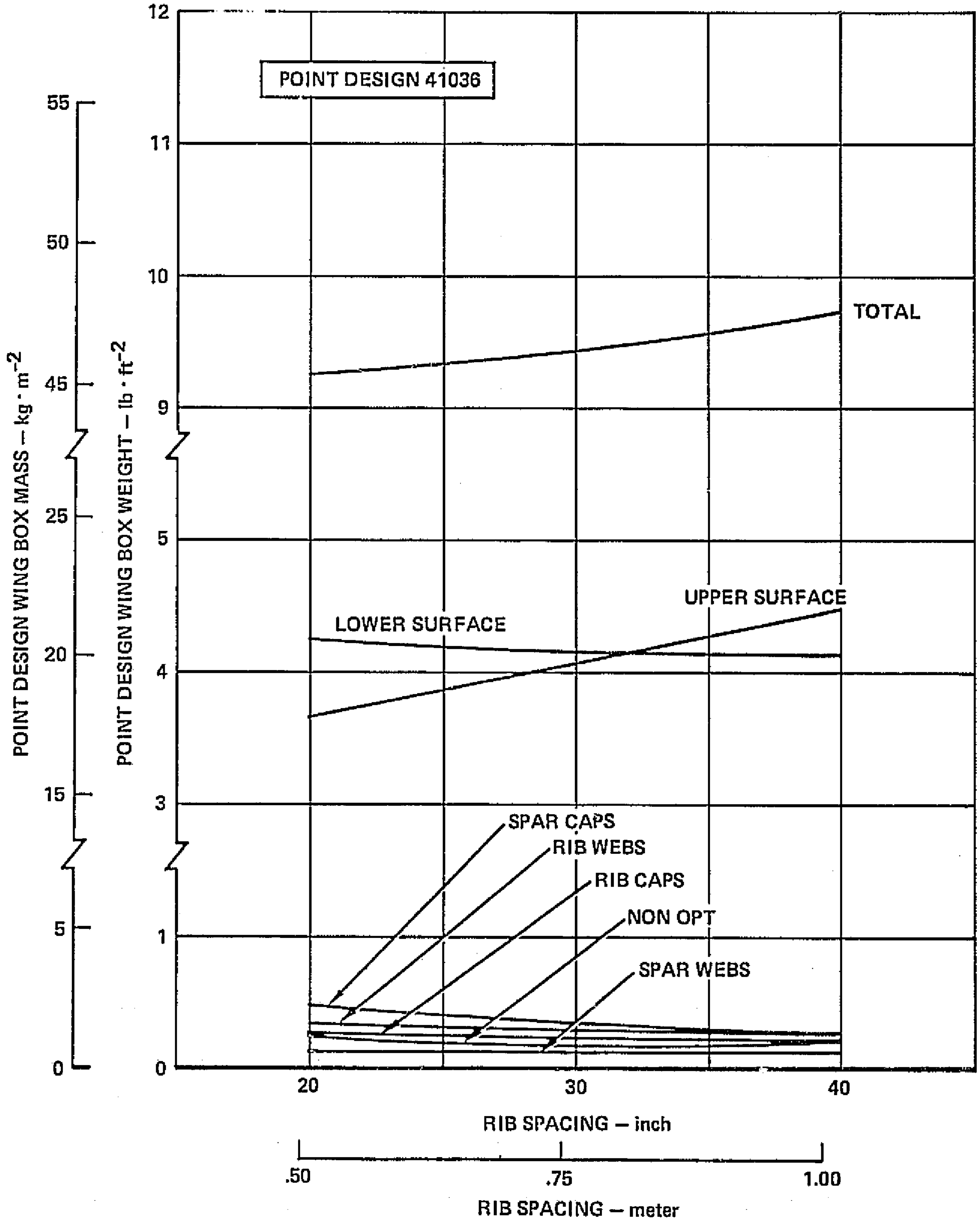


Figure 12-24. Optimum Rib Spacing for Spanwise Wing Arrangement at Point Design Region 41036

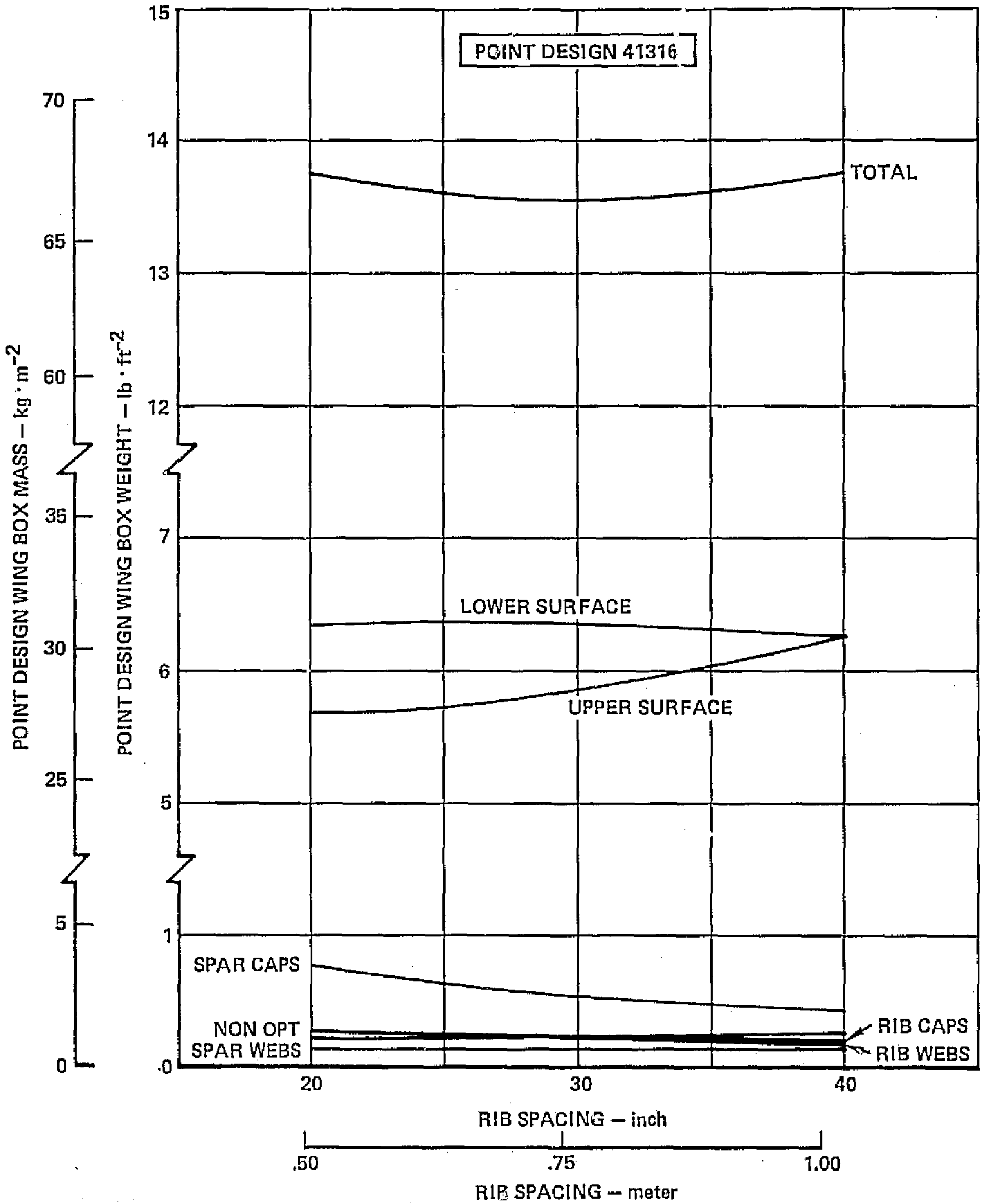


Figure 12-25. Optimum Rib Spacing for Spanwise Wing Arrangemr
Point Design Region 41316

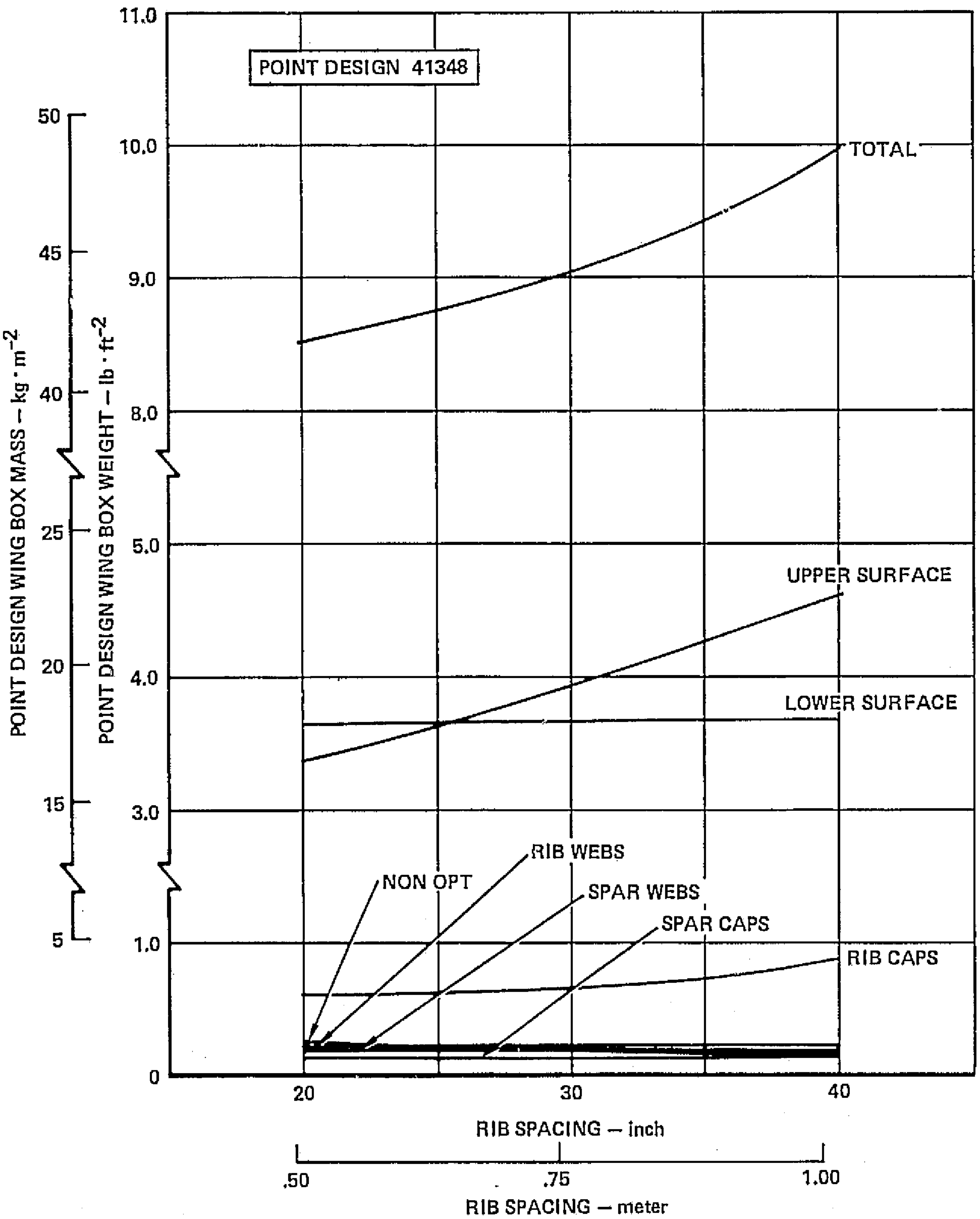


Figure 12-26. Optimum Rib Spacing for Spanwise Wing Arrangement at Point Design Region 41348

lower surface panels with a maximum increment of 1.0 lb/sq.ft. indicated for the 20-inch rib spacing designs, which decreases to 0.20 lb/sq.ft. for the 40-inch design. As with the previously discussed point design regions, the weight of the substructure components vary slightly with rib spacing with no component weighing more than 1.0 lb/sq.ft.

The unit box weight for point design region 41036 representative of the structure located outboard on the wing aft box, are presented in Figure 12-24. A curve with a slight positive slope defines the total box weight with a minimum value of 9.3 lb/sq.ft. occurring for the 20-inch rib spacing design and a maximum value of 9.7 lb/sq.ft. for the 40-inch design. A maximum weight increment of only 0.60 lb/sq.ft. is indicated between the surface panels for the 20-inch spacing design where the heavier lower surface panel weighs 4.2 lb/sq.ft. Identical surface panel weights (zero weight increment) of approximately 4.1 lb/sq.ft. are noted for a 30-inch rib spacing design. For the 40-inch rib spacing designs, the upper surface panel is the heaviest panel and is approximately 0.4 lb/sq.ft. heavier than the corresponding lower surface panel. All substructure components have unit weights less than 0.50 lb/sq.ft. with the spar caps having the maximum values at all rib spacings.

A symmetrical total weight curve is noted in Figure 12-25 for point design region 41316. This curve shows a minimum-weight design of 13.5 lb/sq.ft. for 30-inch rib spacing and identical values of 13.7 lb/sq.ft. for the 20- and 40-inch designs. The predominately-tension designed lower surface panels are heavier than the corresponding designs for the upper surface panels. The exception being the 40-inch rib spacing designs where the surface panels have identical unit weights of 6.25 lb/sq.ft. The heaviest substructure components are the spar caps which have a maximum value of 0.75 lb/sq.ft. for the 20-inch rib spacing design.

The last point design region included in the Detailed Concept Analysis is the mid-span wing tip region 41348. The results of this analysis are presented in Figure 12-26 where the total weight curve indicate the least-weight design occurs for the lowest rib spacing investigated, 20-inches. A unit box weight of 8.5 lb/sq.ft. is noted for this design. The predominately compression designed upper surface panels are heavier than the lower surface panels for designs with rib spacings greater than approximately 25-inches. At this rib spacing, both panels weigh approximately 3.66 lb/sq.ft. The rib caps are the heaviest weight substructure component having a unit weight of 0.90 lb/sq.ft. for the 40-inch rib spacing design.

MONOCOQUE WING ARRANGEMENTS - TASK I

The monocoque (biaxially stiffened) panel concepts were subjected to the same stages of analysis as the uniaxial stiffened panel arrangements, i.e., an Initial Screening and a Detail Concept Analysis. In addition, an additional analysis was conducted at the start of the initial screening to ascertain the minimum weight panel proportions (aspect ratio) prior to screening the candidate concepts.

The two candidate panel concepts are shown in Figure 12-27 and include the honeycomb core and truss core sandwich concepts. In addition this figure contains a sketch of typical monocoque arrangement wing box segment depicting the biaxially stiffened surface panels and related substructure.

The fabrication limits for the monocoque panels and closures are contained in Figure 12-28. These limits include the thickness constraints imposed on the face sheets due to foreign object damage (F.O.D.); which were: .020-inch for the lower surface and .015-inch for the upper surface exposed skins.

The point designs environments for the critical flight conditions are presented in Table 12-27. The critical flight conditions for the Task I analysis were conditions 20 and 31. Condition 20, start of cruise, being the most severe environment for the forward wing box region 40322 with the symmetric flight condition at Mach. 1.25, condition 31, being the most critical for the aft box and wing tip regions.

Monocoque Initial Screening

The candidate monocoque wing panel concepts are biaxially stiffened panels with the two most efficient designs being investigated in the initial screening analysis; these concepts were:

- Honeycomb core sandwich
- Truss core sandwich

The initial screening analysis was conducted at the three point design regions shown in Figure 12-3 using the associated critical point design environments presented in Table 12-27.

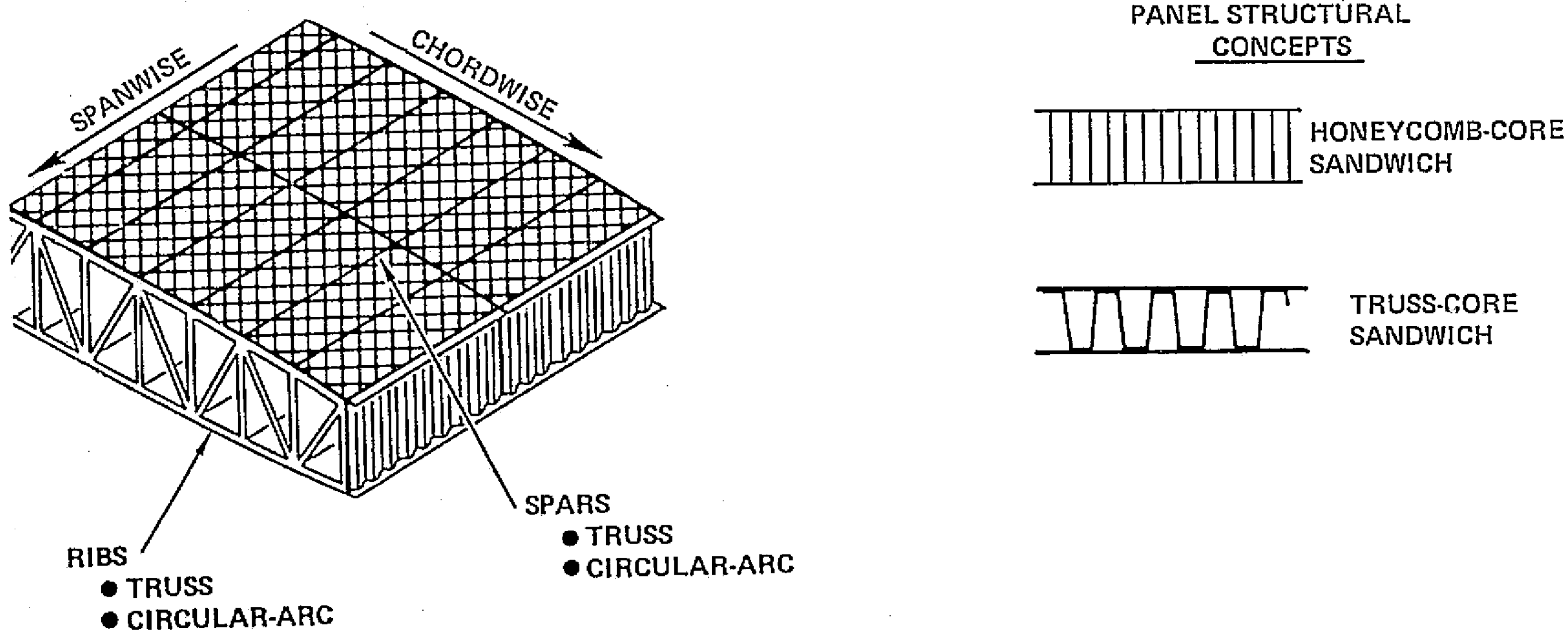


Figure 12-27. Monocoque Stiffened Wing Structural Arrangements

CONCEPT	FACE MATERIAL	CORE MATERIAL	FACE SKIN GAGE (IN.)		SHEET SIZE (IN.)	PANEL SIZE (MAX.)	MATERIAL CONDITION		CORE		DEPTH	BRAZE ALLOY	CORE		FACE STEP THICKNESS		CONTOUR CAPABILITY	COMMENTS	CONCLUSION
							BEFORE BRAZE	BRAZE ALLOY	GAGE (IN.)	CELL SIZE (IN.)			DENSITY	TAPER	CHEM MILL (IN.)	TAPERED SHEET*			
 BRAZED HONEYCOMB	Ti-6Al-4V ANNEALED	Ti-3Al-2.5V	MIN MAX	.010 .008 SPECIAL .030 TO .375 .375	36 x 144 24 x 144 60 x 200 90 x 200	43 IN. X 40 FT (WELD SHEETS)	ANNEALED		.002 .006	1/8 TO 1/2 1/8 TO 1/2	0 TO 4 1/2 IN.	3003 Al	CAN VARY DENSITY AND KEEP SHEAR CONTINUITY	NO PRACTICAL LIMIT UPPER LIMIT CORE COLLAPSE	.010 MIN.	NOT AVAIL-ABLE *MILL ROLLED	NO PRACTICAL LIMIT EXCEPT BRAZE GRAVITY FLOW & STARVATION	BRAZE ALLOY AND ANNEALED CONDITION COMPATIBLE 1240°F BRAZE	LIMITED TO Ti-6Al-4V ANNEALED FACES
	BETA ALLOY (STA)	Ti-3Al-2.5V	MIN MAX	.010 .020 TO .375 .375	24 x 144 60 x 200 96 x 340	68 IN. X 40 FT. (WELD SHEETS)	S.T.	STA REQUIRES EXACT AGEING WITH BRAZING PROCESS AGE 1000°F BRAZE 1240°F	AS ABOVE			NOT RE-SOLVED	AS ABOVE					INITIAL COLD FORMING AN ADVANTAGE METHOD OF AGING AFTER BRAZING NOT RESOLVED. BRAZE ALLOY TO AVOID DIFFUSION NOT RESOLVED.	CONDITION OF BETA ALLOY AFTER AGING UNKNOWN
 STRESSKIN DIFFUSION BONDED	Ti-6Al-4V ANNEALED	Ti-3Al-2.5V	MIN MAX	.010 .008 SPECIAL .063 PRESENT .125 (PROJECTED)	AS ABOVE	4 FT. X 13 FT. (PRESENT) 80 IN. X 24 FT. (PROJECTED)	ANNEALED		.0025 .005	** 1/4 CELL 1/2 THICK 3/8 CELL 2 IN. THICK	0.50 IN. 2.0 IN.	NONE	CAN VARY INDIVIDUAL RIBBON THICKNESS CONSTANT THICKNESS OVER FULL LENGTH OF RIBBON	INDIVIDUAL PANEL CONSTANT THICKNESS	.010 MIN	NOT FEASIBLE	HOT VACUUM AND STRETCH FORMING NO PRACTICAL LIMIT	SHEAR TIE DIFFICULT; BRAZED DOUBLERS MUST BE EXTERNAL; AFTER DIFFUSION BOND SURFACE HAS BUILT IN K, DUE TO INITIAL TACK WELD	LIMITED TO Ti-6Al-4V FACES CONDITION OF BETA ALLOY AFTER DIFFUSION BONDING UNKNOWN
 CORRUGATED SANDWICH	Ti-6Al-4V ANNEALED	Ti-6Al-4V ANNEALED	.010 MIN. .003 SPECIAL		AS ABOVE	APPROX. 80 IN. X 24 FT. DEPENDS ON MFG. PROCESS	ANNEALED		.008		—	3003 Al	NOT APPLICABLE	LIMITED TO STRAIGHT LINE ELEMENTS	.010 MIN	NOT AVAIL-ABLE	FORMABLE IF SPOT OR EB WELDED OR DIFF. BONDED	FORMING DIFFICULT	LIMITED TO Ti-6Al-4V (BRAZED) BETA ALLOY SPOT, EB WELDED OR DIFF. BONDED

**3/16 CELL UNDER DEVELOPMENT

Figure 12-28. Fabrication Limits - Monocoque Stiffened Surface Panels

TABLE 12-27. CRITICAL WING POINT DESIGN ENVIRONMENT, MONOCOQUE ARRANGEMENT – TASK II

CONDITION 20 : (START OF CRUISE); MACH NO. = 2.7; n_z = 2.5

WEIGHT = 860 X 10^3 LB.

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION											
			40236		41036		41316		40536		41348		40322	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	N_x	LB/IN	-1144	1144	-1446	1446	-684	684	-1885	1885.	-45	45	-213	213
	N_y	LB/IN	-7157	7157	-3558	3558	-7901	7901	-7034	7034	-4370	4370	-463	463
	N_{xy}	LB/IN	-1095	1095	-1968	1968	1402	-1402	-2508	2508	1102	-1102	-235	135
THERMAL STRAIN	ϵ_x	IN/IN	-824×10^{-6}	824×10^{-6}	1×10^{-6}	-1×10^{-6}	7×10^{-6}	-7×10^{-6}	-144×10^{-6}	114×10^{-6}	0.3×10^{-6}	-0.3×10^{-6}	-581×10^{-6}	581×10^{-6}
	ϵ_y	IN/IN	-820×10^{-6}	820×10^{-6}	8×10^{-6}	-8×10^{-6}	15×10^{-6}	-15×10^{-6}	129×10^{-6}	-129×10^{-6}	0.38×10^{-6}	0.38×10^{-6}	-538×10^{-6}	538×10^{-6}
	ϵ_{xy}	IN/IN	76×10^{-6}	-76×10^{-6}	-18×10^{-6}	18×10^{-6}	88×10^{-6}	-88×10^{-6}	-144×10^{-6}	144×10^{-6}	-0.1×10^{-6}	0.1×10^{-6}	58×10^{-6}	-58×10^{-6}
PRESSURE	AERO	PSI	-1.70	-.74	-1.25	.38	-1.65	1.00	-1.47	-.36	-1.29	1.04	-1.67	-.17
	FUEL	PSI	-6.42	-7.84	0	0	0	0	-6.00	-7.11	0	0	-6.30	-9.30
	NET	PSI	-8.12	-8.58	-1.25	0.38	-1.65	1.00	-7.47	-7.47	-1.29	1.04	[illegible]	-9.4
TEMPERATURE	T_{AV}	°F	211	205	344	330	313	318	213	207	355	340	246	216
	ΔT	°F	-228	-237	-35	-70	-99	-84	-233	-241	-39	-57	-120	-251

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

CONDITION 31 : MACH NO. = 1.25: n_z = 2.5

WEIGHT = 690 X 10^3 LB.

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION											
			40236		41036		41316		40536		41348		40322	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	N_x	LB/IN	-1193	1193	-2219	2219	-1587	1587	-3171	3171	-1190	1190	51	-51
	N_y	LB/IN	-11638	11638	-6423	6423	-12183	12183	-11404	11404	-7263	7263	-529	529
	N_{xy}	LB/IN	2099	2099	3209	3209	3310	3310	4645	4645	3355	3355	191	191
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	PSI	-3.03	-1.20	-1.27	0.11	-4.98	-.26	-1.27	-.26	-5.05	0.96	-1.47	.05
	FUEL	PSI	-5.93	-8.94	0	0	0	0	-5.67	-8.03	0	0	-6.86	-9.00
	NET	PSI	-8.96	-10.14	-1.27	0.11	-4.98	-0.26	-6.94	-8.29	-5.05	0.96	-8.33	-8.95
TEMPERATURE	T_{AV}	°F	146	147	208	194	178	172	147	148	210	197	169	156
	ΔT	°F	-129	-138	-40	-40	-98	-90	-131	-135	-49	-74	-105	-153

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

The rationale used for evaluating the monocoque panel concepts during the initial screening was:

(1) To ascertain the minimum weight panel proportions by conducting an aspect ratio study using a representative panel concept, honeycomb core sandwich. This included evaluating multispar and multirib designs on a weight bases for panel and wing box designs.

(2) Then conduct a weight/strength analysis for each candidate panel concept using the minimum-weight panel proportions determined from the aspect ratio study.

(3) Compare the results of the above panel analysis and select the most promising concept for further in-depth analysis in the Detailed Concepts Analysis.

The analysis conducted using the above procedure is described in the following text.

Aspect Ratio Study - For this study, various aspect ratios were investigated for multispar and multirib honeycomb core sandwich panel designs. The panel orientation and the general dimensioning associated with these arrangements are shown in Figure 12-29.

Multispar Arrangement - Variable spar spacings of 20-inches, 30-inches, and 40-inches were used for each point design analysis. A constant rib spacing of 60-inches was used for regions 40536 and 41348 with a 130-inch rib spacing selected for the lightly loaded forward box region 40322. For direct comparison between general types of load carrying panels, the rib and spar spacing selected for the multispar arrangement are identical with those selected for the chordwise stiffened panel concepts.

The results of the basic panel sizing for the multispar arrangement are shown in Table 12-28 and includes the panel proportions, cross sectional dimensions, and the weight data. The panel aspect ratio ($L_{p,x}/L_{p,y}$) ranged from 0.33 to 0.67 for regions 40536 and 41348, and .15 to .31 for region 40322. Panel heights (h) varied from approximately .25- to .50-inch, thicknesses from .011-inch to .087-inch, and cell size from .17-inch to .50-inch. A minimum core foil thickness of .002-inch was maintained for all designs. With reference to Table 12-28, the panel equivalent thickness ($\bar{t}$) and unit weight (w) includes the core. In addition,

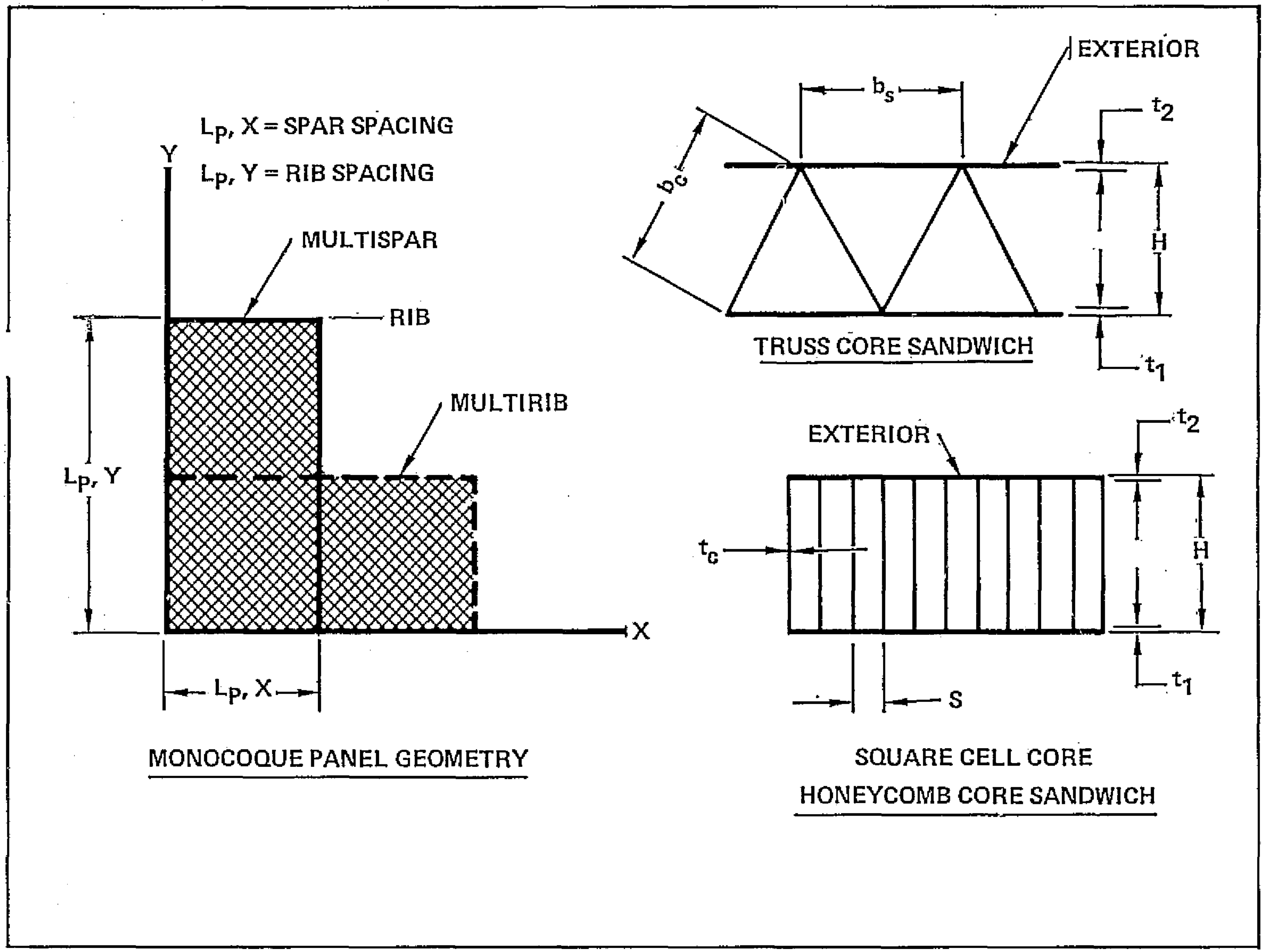


Figure 12-29. Panel Spacing and Cross Section Geometry for the Monocoque Arrangement Aspect Ratio Study

TABLE 12-28. PANEL GEOMETRY AND WEIGHT FOR THE HONEYCOMB CORE SANDWICH PANEL MULTISPAR ARRANGEMENT - ASPECT RATIO STUDY

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
RIB SPACING	(m)	3.30	3.30	3.30	3.30	3.30	3.30	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52
	(in.)	130	130	130	130	130	130	60	60	60	60	60	60	60	60	60	60	60	60
ASPECT RATIO		0.15	0.23	0.31	0.15	0.23	0.31	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67
DIMENSIONS:																			
H	(cm)	1.577	2.530	3.713	2.101	2.652	5.184	2.126	3.233	3.762	0.737	1.153	1.984	1.826	2.647	3.279	0.561	0.917	1.156
	(in.)	0.621	0.996	1.462	0.827	1.044	2.041	0.837	1.273	1.481	0.290	0.454	0.781	0.719	1.042	1.291	0.221	0.361	0.455
t_1	(cm)	0.038	0.046	0.058	0.028	0.038	0.038	0.135	0.132	0.127	0.193	0.193	0.221	0.089	0.091	0.099	0.119	0.178	0.137
	(in.)	0.015	0.018	0.023	0.011	0.015	0.015	0.053	0.052	0.050	0.076	0.076	0.087	0.035	0.036	0.039	0.047	0.070	0.054
t_2	(cm)	0.038	0.038	0.038	0.051	0.051	0.051	0.132	0.130	0.127	0.155	0.160	0.135	0.097	0.097	0.102	0.112	0.051	0.099
	(in.)	0.015	0.015	0.015	0.020	0.020	0.020	0.052	0.051	0.050	0.061	0.063	0.053	0.038	0.038	0.040	0.044	0.020	0.039
t_c	(cm)	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005
	(in.)	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002
S	(cm)	0.597	0.716	0.795	1.270	1.270	1.270	0.655	0.470	0.424	1.270	1.270	1.270	0.808	0.655	0.737	1.270	1.270	1.270
	(in.)	0.235	0.282	0.313	0.500	0.500	0.500	0.258	0.185	0.167	0.500	0.500	0.500	0.318	0.258	0.290	0.500	0.500	0.500
MASS DATA:																			
$\overline{t}$	(cm)	0.102	0.119	0.142	0.097	0.109	0.130	0.297	0.333	0.353	0.353	0.361	0.368	0.208	0.226	0.244	0.234	0.234	0.241
	(in.)	0.040	0.047	0.056	0.038	0.043	0.051	0.117	0.131	0.139	0.139	0.142	0.145	0.082	0.089	0.096	0.092	0.092	0.095
W	($kg \cdot m^{-2}$)	4.511	5.263	6.289	4.248	4.863	5.737	13.124	14.740	15.682	15.634	16.000	16.346	9.169	10.009	10.785	10.336	10.370	10.658
	($lb \cdot ft^{-2}$)	0.924	1.078	1.288	0.870	0.996	1.175	2.688	3.019	3.212	3.202	3.277	3.348	1.878	2.050	2.209	2.117	2.124	2.183
W_c	($kg \cdot m^{-2}$)	1.040	1.538	2.051	0.718	0.908	1.850	1.279	3.095	4.458	0.137	0.283	0.576	0.913	1.694	1.880	0.117	0.244	0.327
	($lb \cdot ft^{-2}$)	0.213	0.315	0.420	0.147	0.186	0.379	0.262	0.634	0.913	0.028	0.058	0.118	0.187	0.347	0.385	0.024	0.050	0.067
ρ_c	($kg \cdot m^{-3}$)	75.431	62.824	56.641	35.433	35.433	35.433	68.880	104.25	127.12	35.433	35.433	35.433	55.632	68.863	61.062	35.433	35.433	35.433
	($lb \cdot ft^{-3}$)	4.709	3.922	3.536	2.212	2.212	2.212	4.300	6.508	7.936	2.212	2.212	2.212	3.473	4.299	3.812	2.212	2.212	2.212
CRITICAL CONDITION		20	20	20	20	20	20	31	31	31	31	31	31	31	31	31	31	31	31

NOTE: (1) ASPECT RATIO = $L_{P,X}/L_{P,Y}$
(2) BRAZE MATERIAL NOT INCLUDED

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the core density (ρ_c) and weight (w_c) are also listed separately. The panel unit weights, includes face sheets and core, range from a minimum weight of 0.90 lb/sq.ft. to a maximum of 3.35 lb/sq.ft.

The weight related to the panel fabrication technique was investigated to obtain a more realistic comparison between arrangements. The weight data for the 3003 aluminum braze alloy used for panel fabrication was obtained from empirical data reported in Reference 3 and is shown in Figure 12-30. Using this data and the basic panel results shown in Table 12-28, the combined panel weights were calculated and are presented in Table 12-29 for the multispar design.

Multirib Arrangement - For this arrangement, variable rib spacings of 20-inches, 30-inches, and 40-inches were used at each point design region with a constant spar spacing of 60-inches. Similar to the panel dimension criteria used on the multispar arrangement, the multirib panel dimensions were selected for direct comparison with the uniaxial spanwise stiffened arrangements.

The results of the panel sizing analysis conducted on the multirib panel designs are summarized in Table 12-30. With reference to this table, the aspect ratio varied from 1.5 for the larger rib spacing to 3.0 for the smaller spacing. The panel heights, face sheet thicknesses, and core cell sizes, ranged from: 0.85-inch to 2.0-inch, 0.012-inch to 0.105-inch, and 0.20-inch to 0.50-inch, respectively. The panel weights, sum of the face sheets and core, varied from approximately 0.90 lb/sq.ft. to 3.50 lb/sq.ft. The braze weights were defined using Figure 12-30 and are included in the panel weight summary for the multirib arrangement presented in Table 12-29.

Aspect Ratio Study Results - Table 12-29 presents both designs for comparison purposes and includes the individual panel, braze material, and combined weight for both the multispar and multirib designs. For clarity in reporting, these values are displayed graphically in Figure 12-31. From a review of this figure, the weight of the multispar panel arrangements were lighter than those of the multirib designs at all point design regions. The exception being at region 40322 where the multirib design has a slightly lower weight, i.e., approximately 1-percent lighter than the multispar design. In addition, the minimum-weight designs for both arrangements are those associated with the smallest spar spacing, 20-inches.

Panel weights for the 20-inch spar spacing multispar designs at regions 40322, 41349, and 40536 are 2.1 lb/sq.ft., 4.3 lb/sq.ft., and 6.2 lb/sq.ft., respectively.

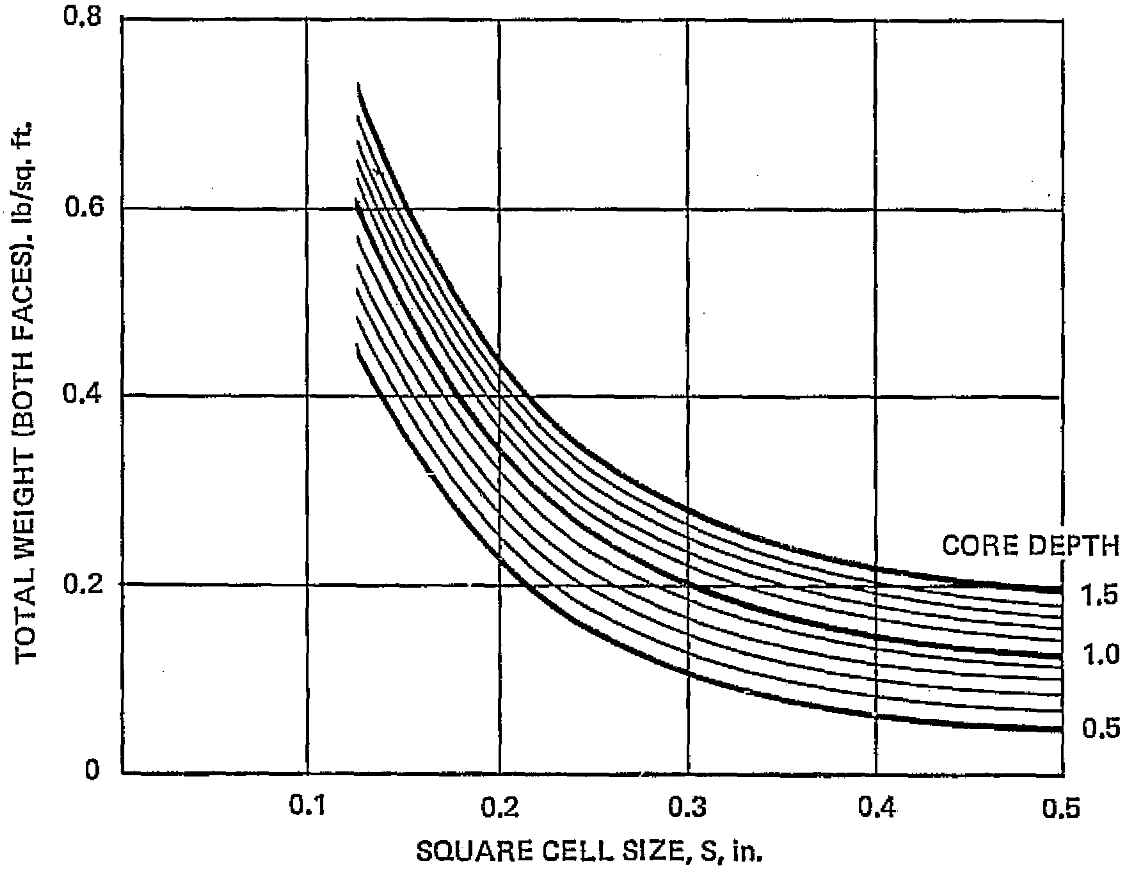


Figure 12-30. Honeycomb Core Sandwich Braze Weight,
3003 Aluminum Braze Alloy

TABLE 12-29. SURFACE PANEL WEIGHT COMPARISON OF THE MULTISPAR AND MULTIRIB DESIGNS - ASPECT RATIO STUDY

ITEM \ WT.	SURFACE PANEL UNIT WEIGHT (LB./SQ. FT.)								
POINT DESIGN REGION	40322			40536			41348		
SPAR SPACING (IN.)	20	30	40	20	30	40	20	30	40
MULTI SPAR DESIGN									
• UPPER SURFACE	(1.14)	(1.31)	(1.61)	(2.92)	(3.49)	(3.77)	(2.02)	(2.31)	(2.47)
PANEL	0.92	1.08	1.29	2.69	3.02	3.21	1.88	2.05	2.21
BRAZE	0.22	0.23	0.32	0.23	0.47	0.56	0.14	0.26	0.26
• LOWER SURFACE	(0.98)	(1.12)	(1.44)	(3.30)	(3.38)	(3.46)	(2.24)	(2.22)	(2.29)
PANEL	0.87	1.00	1.18	3.20	3.28	3.35	2.12	2.12	2.18
BRAZE	0.11	0.12	0.26	0.10	0.10	0.11	0.12	0.10	0.11
• TOTAL $\sum$	(2.12)	(2.43)	(3.05)	(6.22)	(6.87)	(7.23)	(4.26)	(4.53)	(4.76)
MULTI RIB DESIGN									
UPPER SURFACE	(1.13)	(1.30)	(1.43)	(3.08)	(3.99)	(4.10)	(2.21)	(2.55)	(2.72)
PANEL	0.93	1.07	1.19	2.78	3.41	3.51	1.99	2.30	2.43
BRAZE	0.20	0.23	0.24	0.30	0.58	0.59	0.22	0.25	0.29
LOWER SURFACE	(0.96)	(1.10)	(1.25)	(3.27)	(3.54)	(3.66)	(2.20)	(2.26)	(2.29)
PANEL	0.86	0.98	1.07	3.26	3.39	3.48	2.10	2.16	2.19
BRAZE	0.10	0.12	0.18	0.01	0.15	0.18	0.10	0.10	0.10
TOTAL $\sum$	(2.09)	(2.40)	(2.68)	(6.35)	(7.53)	(7.76)	(4.41)	(4.81)	(5.01)

TABLE 12-30. PANEL GEOMETRY AND WEIGHT FOR THE MULTIRIB DESIGN HONEYCOMB CORE SANDWICH PANELS, MONOCOQUE ARRANGEMENT - ASPECT RATIO STUDY

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52
	(in.)	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
ASPECT RATIO		3.0	2.0	1.5	3.0	2.0	1.5	3.0	2.0	1.5	3.0	2.0	1.5	3.0	2.0	1.5	3.0	2.0	1.5
DIMENSIONS:																			
H	(cm)	1.689	2.421	3.284	1.831	2.418	3.810	3.239	5.034	5.438	2.164	2.850	3.503	2.761	3.678	4.199	1.374	1.654	1.593
	(in.)	0.665	0.953	1.293	0.721	0.952	1.500	1.275	1.982	2.141	0.852	1.122	1.379	1.087	1.448	1.653	0.541	0.651	0.627
t_1	(cm)	0.038	0.046	0.051	0.030	0.038	0.036	0.130	0.132	0.132	0.079	0.102	0.130	0.097	0.122	0.130	0.102	0.109	0.130
	(in.)	0.015	0.018	0.020	0.012	0.015	0.014	0.051	0.052	0.052	0.031	0.040	0.051	0.038	0.048	0.051	0.040	0.043	0.051
t_2	(cm)	0.038	0.038	0.038	0.051	0.051	0.051	0.130	0.132	0.132	0.267	0.254	0.229	0.086	0.089	0.089	0.122	0.117	0.102
	(in.)	0.015	0.015	0.015	0.020	0.020	0.020	0.051	0.052	0.052	0.105	0.100	0.090	0.034	0.035	0.035	0.048	0.046	0.040
t_c	(cm)	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005
	(in.)	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002
S	(cm)	0.622	0.721	0.754	1.270	1.270	1.270	0.638	0.462	0.460	1.270	1.262	1.270	0.714	0.841	0.787	1.270	1.270	1.270
	(in.)	0.245	0.284	0.297	0.500	0.500	0.500	0.251	0.182	0.181	0.500	0.497	0.500	0.281	0.331	0.310	0.500	0.500	0.500
MASS DATA:																			
$\bar{t}$	(cm)	0.102	0.117	0.132	0.094	0.109	0.117	0.307	0.376	0.386	0.361	0.373	0.384	0.218	0.254	0.269	0.234	0.239	0.241
	(in.)	0.040	0.046	0.052	0.037	0.043	0.046	0.121	0.148	0.152	0.142	0.147	0.151	0.086	0.100	0.106	0.092	0.094	0.095
W	(kg·m^{-2})	4.550	5.210	5.800	4.179	4.809	5.205	13.598	16.654	17.147	15.922	16.561	16.976	9.721	11.230	11.869	10.258	10.551	10.678
	(lb·ft^{2})	0.932	1.067	1.188	0.856	0.985	1.066	2.785	3.411	3.512	3.261	3.392	3.477	1.991	2.300	2.431	2.101	2.161	2.187
W_c	(kg·m^{-3})	1.167	1.455	1.826	0.620	0.825	1.318	2.099	5.009	5.429	0.644	0.889	1.113	1.626	1.855	2.275	0.410	0.508	0.483
	(lb·ft^{-3})	0.239	0.298	0.374	0.127	0.169	0.270	0.430	1.026	1.112	0.132	0.182	0.228	0.333	0.380	0.466	0.084	0.104	0.099
ρ_c	(kg·m^{-2})	72.419	62.200	59.749	35.433	35.433	35.433	70.561	104.94	104.97	35.417	35.673	35.449	63.129	53.486	57.154	35.433	35.433	35.433
	(lb·ft)	4.521	3.883	3.730	2.212	2.212	2.212	4.405	6.551	6.553	2.211	2.227	2.213	3.941	3.339	3.568	2.212	2.212	2.212
CRITICAL CONDITION		20	20	20	20	20	20	31	31	31	31	31	31	31	31	31	31	31	31

NOTE: (1) ASPECT RATIO = $L_{P,X}/L_{P,Y}$
(2) BRAZE MATERIAL NOT INCLUDED

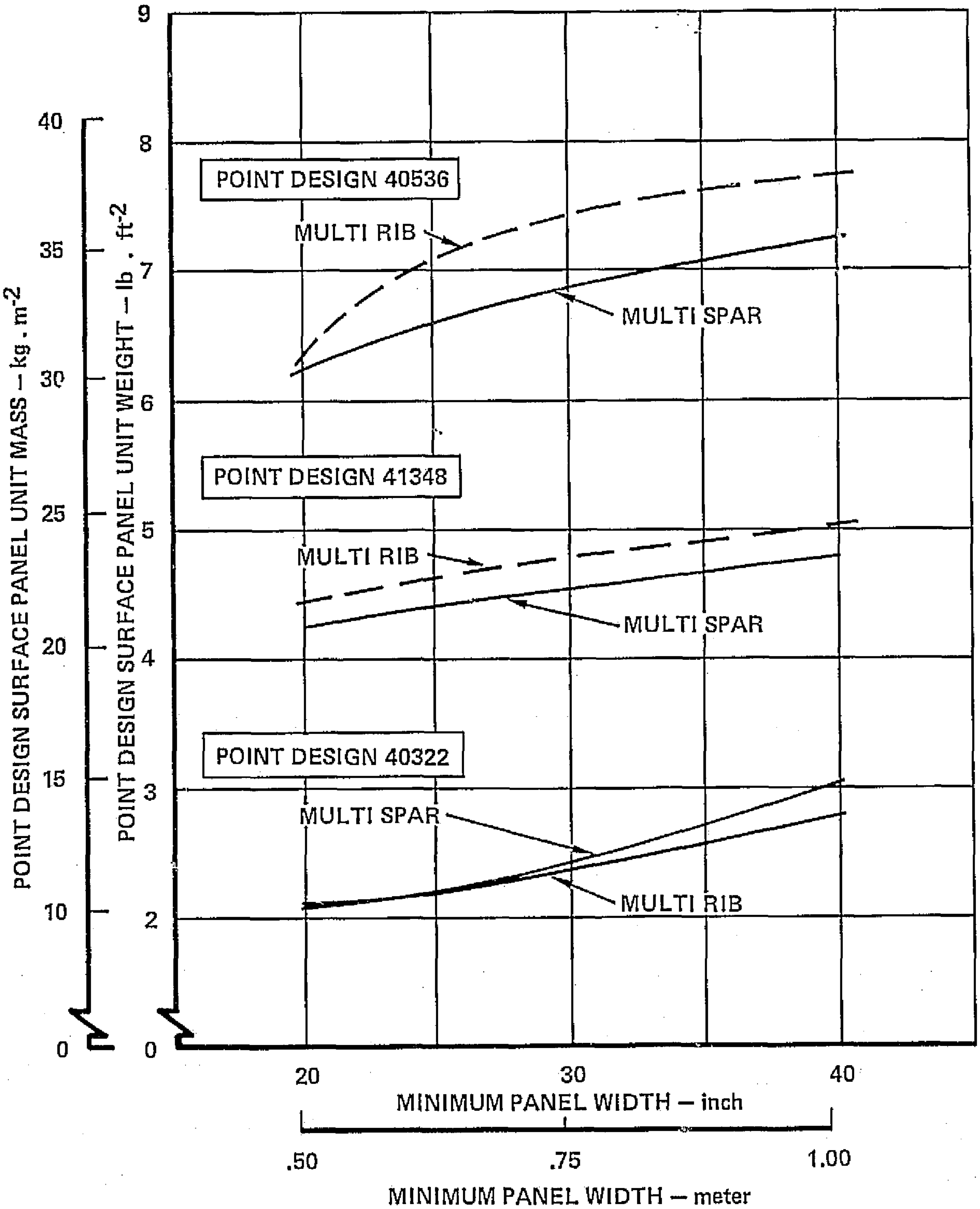


Figure 12-31. Comparison of Surface Panel Weight for the Multispar and Multirib Designs

In general, the largest difference in weight between the multispar and the multirib arrangements occur at Region 40536 where a difference of 0.66 lb/sq.ft. is noted for the 30-inch spacing designs.

Conversely, no appreciable weight difference is indicated between the 20-inch spacing multispar and multirib designs at any of these regions. A maximum weight difference of approximately 0.20 lb/sq.ft. is noted at Region 41348 where the lightest design is the multispar design.

The panel aspect ratio study indicates the multispar panel arrangement results in the lightest weight designs for the larger panel widths, but the data is inconclusive for the smaller panel widths.

To establish the weight trends for the multispar and multirib designs at the smaller panel widths, the aspect ratio study was extended to include the weight of the associated substructure. In addition to the panels sized for the panel aspect ratio study, the weights attributed to the rib and spar caps, rib and spar webs, and non-optimum factors were included in the wing box aspect ratio study. For this study, only the results will be presented since a thorough description of the substructure is included in the following Detailed Concepts evaluation.

The results of the unit box weight study are summarized in Tables 12-31 and 12-32 for the multispar and multirib arrangements and include the unit weight for each component and the total box weight. As with the panel aspect ratio study, point design regions 40322, 40536, and 41348 were used for this analysis.

For an interpretation of these results the point design box weights are displayed graphically in Figure 12-32. With respect to this figure, the multispar arrangement affords a lighter weight design for all panel widths. The exception being the 35-inch or greater multirib designs at Region 40322 which are lighter than the corresponding multispar designs. The minimum-weight multispar arrangements are the 20-inch spar spacing designs which weigh 4.5, 5.8, and 8.7 lb/sq.ft. for Regions 40322 41348, and 40536 respectively.

In conclusion, the multispar panel arrangements affords the minimum-weight designs from both a panel and wing box segment standpoint. In addition, the inclusion of

TABLE 12-31. DETAIL WING WEIGHTS FOR THE HONEYCOMB SANDWICH PANEL MULTISPAR ARRANGEMENT - ASPECT RATIO STUDY

POINT DESIGN REGION	40322			40536			41348		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	1.144	1.308	1.608	2.918	3.489	3.772	2.018	2.310	2.469
LOWER	0.980	1.126	1.445	3.302	3.377	3.458	2.241	2.224	2.288
Σ	(2.124)	(2.434)	(3.053)	(6.220)	(6.866)	(7.230)	(4.259)	(4.534)	(4.757)
RIB WEBS									
BULKHEAD	0.241	0.241	0.241	0.244	0.244	0.244	0.100	0.100	0.100
TRUSS	0.198	0.198	0.198	0.229	0.229	0.229	—	—	—
Σ	(0.439)	(0.439)	(0.439)	(0.473)	(0.473)	(0.473)	(0.100)	(0.100)	(0.100)
SPAR WEBS									
BULKHEAD	0.355	0.343	0.352	0.245	0.285	0.321	0.401	0.301	0.251
TRUSS	0.339	0.194	0.121	0.590	0.389	0.326	—	—	—
Σ	(0.694)	(0.537)	(0.473)	(0.835)	(0.674)	(0.647)	(0.401)	(0.301)	(0.251)
RIB CAPS									
UPPER	0.120	0.140	0.158	0.127	0.143	0.150	0.127	0.146	0.154
LOWER	0.133	0.138	0.193	0.100	0.106	0.127	0.099	0.106	0.113
Σ	(0.253)	(0.278)	(0.351)	(0.227)	(0.249)	(0.277)	(0.226)	(0.252)	(0.267)
SPAR CAPS									
UPPER	0.401	0.307	0.268	0.418	0.311	0.242	0.376	0.289	0.227
LOWER	0.445	0.316	0.325	0.332	0.236	0.206	0.290	0.208	0.167
Σ	(0.846)	(0.623)	(0.591)	(0.750)	(0.547)	(0.448)	(0.666)	(0.497)	(0.394)
NON-OPTIMUM									
MECH. FAST.	0.050	0.040	0.030	0.050	0.040	0.030	0.050	0.040	0.030
WEB INTERS.	0.113	0.098	0.091	0.131	0.115	0.112	0.050	0.040	0.035
Σ	(0.163)	(0.138)	(0.121)	(0.181)	(0.155)	(0.142)	(0.100)	(0.080)	(0.065)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	4.519	4.449	5.028	8.686	8.964	9.217	5.752	5.764	5.834

TABLE 12-32. DETAIL WING WEIGHTS FOR THE HONEYCOMB SANDWICH PANEL MULTIRIB ARRANGEMENT - ASPECT RATIO STUDY

POINT DESIGN REGION	40322			40536			41348		
RIB SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	1.132	1.297	1.428	3.085	3.991	4.102	2.211	2.550	2.721
LOWER	0.956	1.105	1.256	3.271	3.542	3.657	2.201	2.261	2.287
Σ	(2.088)	(2.402)	(2.684)	(6.356)	(7.533)	(7.759)	(4.412)	(4.81)	(5.008)
RIB WEBS									
BULKHEAD	0.446	0.446	0.454	0.292	0.333	0.430	0.250	0.167	0.133
TRUSS	0.300	0.164	0.115	0.541	0.330	0.245	—	—	—
Σ	(0.746)	(0.610)	(0.569)	(0.833)	(0.663)	(0.675)	(0.250)	(0.167)	(0.133)
SPAR WEBS									
BULKHEAD	0.352	0.352	0.352	0.388	0.388	0.388	0.251	0.251	0.251
TRUSS	0.048	0.048	0.048	0.137	0.137	0.137	—	—	—
Σ	(0.400)	(0.400)	(0.400)	(0.525)	(0.525)	(0.525)	(0.251)	(0.251)	(0.251)
RIB CAPS									
UPPER	0.406	0.296	0.240	0.491	0.364	0.281	0.447	0.504	0.532
LOWER	0.421	0.300	0.268	0.333	0.296	0.246	0.353	0.373	0.366
Σ	(0.827)	(0.596)	(0.508)	(0.824)	(0.660)	(0.527)	(0.800)	(0.877)	(0.898)
SPAR CAPS									
UPPER	0.132	0.146	0.161	0.157	0.189	0.190	0.198	0.223	0.235
LOWER	0.137	0.149	0.180	0.138	0.153	0.166	0.157	0.165	0.162
Σ	(0.269)	(0.295)	(0.341)	(0.295)	(0.342)	(0.356)	(0.355)	(0.388)	(0.397)
NON-OPTIMUM									
MECH. FAST.	0.180	0.170	0.160	0.200	0.190	0.180	0.200	0.190	0.180
WEB INTERS.	0.115	0.101	0.097	0.136	0.119	0.120	0.050	0.042	0.038
Σ	(0.295)	(0.271)	(0.257)	(0.336)	(0.309)	(0.300)	(0.250)	(0.232)	(0.218)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	4.625	4.574	4.759	9.169	10.032	10.142	6.318	6.726	6.905

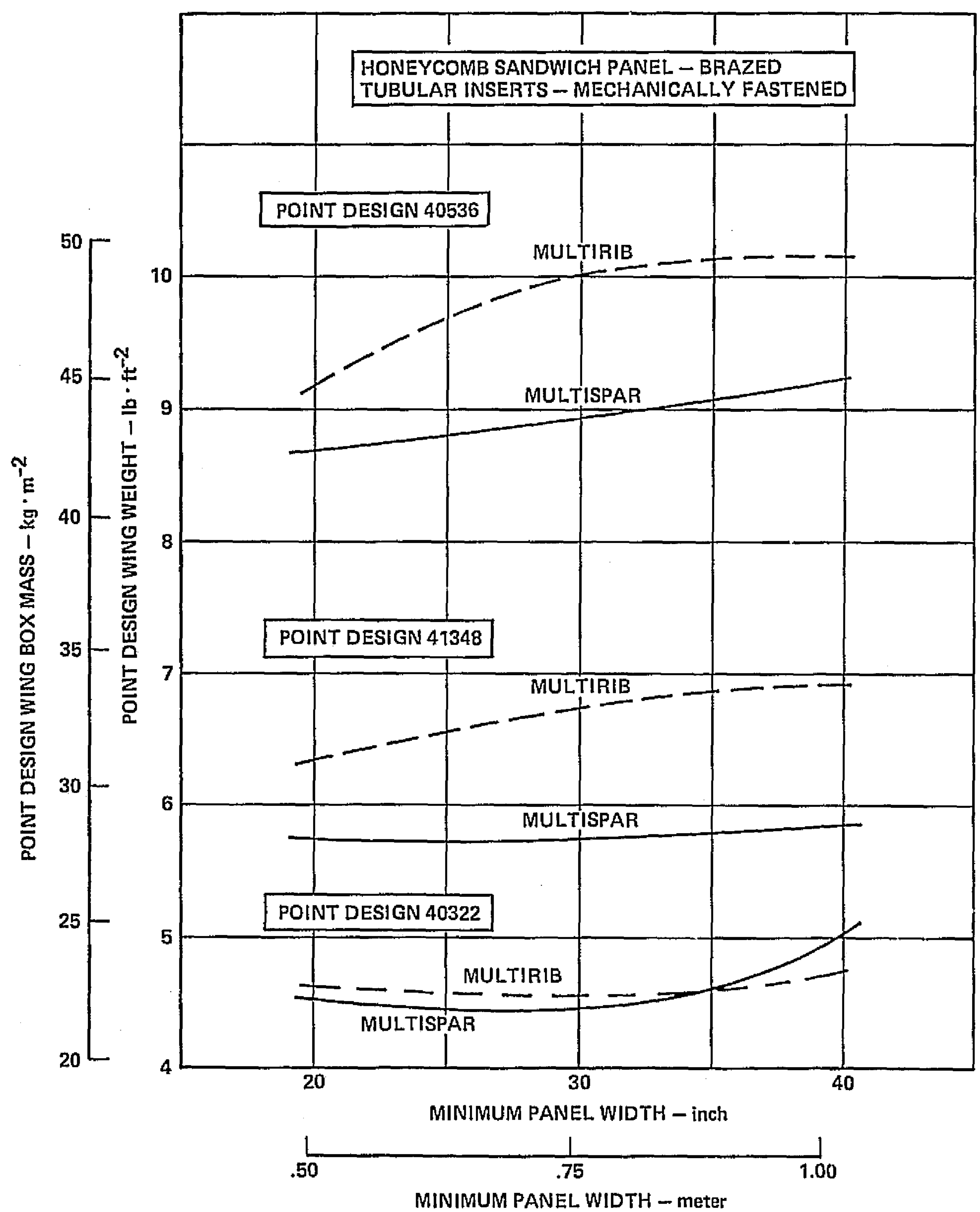


Figure 12-32. Comparison of Wing Box Weight for the Multispar and Multirib Designs

substructure in the analysis resulted in a larger variation between the arrangements and provides a much clearer definition of the minimum weight arrangement.

Panel Screening - A structural analysis was conducted on the two candidate monocoque panel concepts (honeycomb-core sandwich and truss-core sandwich) to define an accurate weight for each concept and select the most promising concept for further study in the Detail Concept Analysis. This screening analysis was conducted at three point design regions using the related monocoque load/temperature environment specified in Table 12-27. In addition, the metallic material used for these concepts was 6Al-4V (ANN.) titanium alloy and the panel proportions were commensurate with the findings of the aspect ratio study, i.e., multispar arrangement. For this arrangement variable spar spacings of 20-inches, 30-inches, and 40-inches were investigated for a constant rib spacing.

Honeycomb-Core Sandwich - Since the material system and applied loads (point design environment) are identical to those used for the aspect ratio study, the panel sizing data calculated for the multispar structural arrangement is also applicable for this analysis. Table 12-28 contains these results which included the basic cross section dimensions and weight data for each design. The total panel weights (combined weight of the basic panel and aluminum braze) for the honeycomb core sandwich concept are presented in Table 12-33 and includes the braze weight as determined from Figure 12-30.

Truss-Core Sandwich - The basic panel sizing results for the truss-core sandwich are presented in Table 12-34 and contains similar cross sectional properties and weight data as shown for the honeycomb core sandwich panels. With respect to this table, minimum panel weights are noted for region 40322 where the weights ranged from a minimum of 1.32 lb/sq.ft. for the lower surface panel with 20-inch spar spacing to a maximum of 2.70 lb/sq.ft. for the upper surface 40-inch spar spacing design. For region 40536, the panel weight varied from approximately 3.2 lb/sq.ft. to 4.5 lb/sq.ft. Similarly, the weight range at point design region 41348 was approximately 2.0 lb/sq.ft. to 3.2 lb/sq.ft. In addition, this table indicates the critical Task I flight condition designing each region, see Section 11 for a description of the flight parameters.

TABLE 12-33. COMPARISON OF THE CANDIDATE MONOCOQUE SURFACE PANEL WEIGHTS - INITIAL SCREENING

ITEM \ WT.	SURFACE PANEL UNIT WEIGHT (LB./SQ. FT.)								
POINT DESIGN REGION	40322			40536			41348		
SPAR SPACING (IN.)	20	30	40	20	30	40	20	30	40
HONEYCOMB CORE SANDWICH									
• UPPER SURFACE	(1.14)	(1.31)	(1.61)	(2.92)	(3.49)	(3.77)	(2.02)	(2.31)	(2.47)
PANEL	0.92	1.08	1.29	2.69	3.02	3.21	1.88	2.05	2.21
FAB. METHOD (BRAZE)	0.22	0.23	0.32	0.23	0.47	0.56	0.14	0.26	0.26
• LOWER SURFACE	(0.98)	(1.12)	(1.44)	(3.30)	(3.38)	(3.46)	(2.24)	(2.22)	(2.29)
PANEL	0.87	1.00	1.18	3.20	3.28	3.35	2.12	2.12	2.18
FAB. METHOD (BRAZE)	0.11	0.12	0.26	0.10	0.10	0.11	0.12	0.10	0.11
• TOTAL $\sum$	(2.12)	(2.43)	(3.05)	(6.22)	(6.87)	(7.23)	(4.26)	(4.53)	(4.76)
TRUSS-CORE SANDWICH									
UPPER SURFACE	(1.61)	(2.00)	(2.70)	(3.16)	(3.73)	(4.46)	(2.15)	(2.70)	(3.25)
PANEL	1.61	2.00	2.70	3.16	3.73	4.46	2.15	2.70	3.23
FAB. METHOD (DIFF. BOND.)	—	—	—	—	—	—	—	—	—
LOWER SURFACE	(1.32)	(1.34)	(1.95)	(3.31)	(3.47)	(3.60)	(2.00)	(2.12)	(2.19)
PANEL	1.32	1.34	1.95	3.31	3.47	3.60	2.00	2.12	2.19
FAB. METHOD (DIFF. BOND)	—	—	—	—	—	—	—	—	—
TOTAL $\sum$	(2.93)	(3.34)	(4.65)	(6.47)	(7.20)	(8.06)	(4.15)	(4.82)	(5.44)

TABLE 12-34. PANEL GEOMETRY AND WEIGHT FOR THE MULTISPAR DESIGN TRUSS - CORE SANDWICH PANELS - INITIAL SCREENING

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
RIB SPACING	(m)	3.30	3.30	3.30	3.30	3.30	3.30	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52
	(in.)	130	130	130	130	130	130	60	60	60	60	60	60	60	60	60	60	60	60
ASPECT RATIO		0.15	0.23	0.31	0.15	0.23	0.31	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67
DIMENSIONS:																			
H	(cm)	1.557	2.515	3.830	1.067	1.148	2.802	2.337	3.094	4.028	0.556	0.871	1.176	1.661	2.380	3.043	0.734	1.204	1.260
	(in.)	0.613	0.990	1.508	0.420	0.452	1.103	0.920	1.218	1.586	0.219	0.343	0.463	0.654	0.937	1.198	0.289	0.474	0.496
t_1	(cm)	0.051	0.064	0.058	0.030	0.033	0.048	0.135	0.147	0.145	0.170	0.180	0.188	0.107	0.119	0.137	0.104	0.117	0.119
	(in.)	0.020	0.025	0.023	0.012	0.013	0.019	0.053	0.058	0.057	0.067	0.071	0.074	0.042	0.047	0.054	0.041	0.046	0.047
t_2	(cm)	0.038	0.051	0.051	0.051	0.051	0.051	0.117	0.119	0.127	0.160	0.165	0.168	0.069	0.081	0.094	0.089	0.086	0.084
	(in.)	0.015	0.020	0.020	0.020	0.020	0.020	0.046	0.047	0.050	0.063	0.065	0.066	0.027	0.032	0.037	0.035	0.034	0.033
t_c	(cm)	0.025	0.025	0.028	0.028	0.025	0.025	0.041	0.048	0.056	0.028	0.025	0.026	0.030	0.036	0.043	0.025	0.025	0.028
	(in.)	0.010	0.010	0.011	0.011	0.010	0.010	0.016	0.019	0.022	0.011	0.010	0.011	0.012	0.014	0.017	0.010	0.010	0.011
b_s	(cm)	0.917	1.207	1.148	0.889	0.968	1.234	2.009	2.047	2.057	1.118	1.361	1.681	1.697	1.737	2.090	3.053	3.485	2.624
	(in.)	0.361	0.475	0.452	0.350	0.381	0.486	0.791	0.806	0.810	0.440	0.536	0.662	0.668	0.684	0.823	1.202	1.372	1.033
b_c	(cm)	1.580	2.532	3.820	1.118	1.207	2.822	2.428	3.132	4.023	0.683	0.975	1.306	1.788	2.441	3.109	1.656	2.062	1.750
	(in.)	0.622	0.997	1.504	0.440	0.475	1.111	0.956	1.233	1.584	0.269	0.384	0.514	0.704	0.961	1.224	0.652	0.812	0.689
θ	(rad)	1.278	1.330	1.421	1.162	1.159	1.351	1.145	1.237	1.312	0.613	0.799	0.871	1.077	1.208	1.229	0.396	0.564	0.724
	(deg)	73.2	76.2	81.4	66.6	66.4	77.4	65.6	70.9	75.2	35.1	45.8	49.9	61.7	69.2	70.4	22.7	32.3	41.5
MASS DATA:																			
$\overline{t}$	(cm)	0.178	0.221	0.297	0.145	0.147	0.213	0.348	0.411	0.490	0.363	0.381	0.396	0.236	0.297	0.356	0.221	0.234	0.241
	(in.)	0.070	0.087	0.117	0.057	0.058	0.084	0.137	0.162	0.193	0.143	0.150	0.156	0.093	0.117	0.140	0.087	0.092	0.095
W	($kg \cdot m^{-2}$)	7.856	9.775	13.193	6.435	6.552	9.501	15.424	18.226	21.756	16.141	16.927	17.596	10.497	13.187	15.751	9.740	10.331	10.678
	($lb \cdot ft^{-2}$)	1.609	2.002	2.702	1.318	1.342	1.946	3.159	3.733	4.456	3.306	3.467	3.604	2.150	2.701	3.226	1.995	2.116	2.187
W_c	($kg \cdot m^{-2}$)	3.881	4.721	8.378	2.856	2.817	5.146	4.292	6.420	9.667	1.553	1.645	1.855	2.759	4.394	5.512	1.221	1.333	1.640
	($lb \cdot ft^{-2}$)	0.795	0.967	1.716	0.585	0.577	1.054	0.879	1.315	1.980	0.318	0.337	0.380	0.565	0.900	1.129	0.250	0.273	0.336
ρ_c	($kg \cdot m^{-3}$)	256.71	192.06	221.84	278.75	254.37	186.94	193.98	216.87	248.46	395.02	235.63	185.86	175.15	192.63	188.22	191.58	120.76	141.59
	($lb \cdot ft^{-3}$)	16.026	11.990	13.849	17.402	15.880	11.670	12.110	13.539	15.511	24.660	14.710	11.603	10.934	12.026	11.750	11.960	7.539	8.839
CRITICAL CONDITION		20	20	20	20	20	20	31	31	31	31	31	31	31	31	31	31	31	31

NOTE: (1) ASPECT RATIO = $L_{P,X}/L_{P,Y}$

(2) PANELS HAVE SPANWISE STIFFENING

The diffusion bonded technique was employed for fabricating the skin-to-core attachment. This process results in no discernable weight penalty for the basic truss-core panel as summarized in Table 12-33.

Monocoque Surface Panel Results - The results of the panel sizing analysis conducted on the candidate monocoque concepts are presented in Table 12-33 and displayed graphically in Figure 12-33. With respect to Figure 12-33, the honeycomb core sandwich concept is the lightest-weight concept at each of the three regions investigated for this study. The exception being region 41348, midspan wing tip location, where the truss-core and honeycomb-core concepts have approximately the same weight (4.2 lb/sq.ft.) for the 20-inch spar spacing design. The least-weight designs for each concept occur at spar spacing of 20-inches. For this spacing, the minimum-weight honeycomb sandwich concept has unit weights of 2.1 lb/sq.ft., 4.2 lb/sq.ft., and 6.2 lb/sq.ft. for region 40322, 41348, and 40536, respectively.

Similar to the chordwise and spanwise initial screening analyses, an additional weight-trend study was conducted where the candidate panel concepts were applied to representative wing box structure. For this analysis, the wing box weight at region 40536 was defined for each panel concept using typical substructure and panel close-out designs. Figure 12-34 contains the close-out designs for the two panel concepts.

The detail wing weights for the two concepts are shown in Table 12-35 and presented graphically in Figure 12-35. From a review of Table 12-35, the predominant weight component for each design are the surface panels with the spar webs and rib webs ranked a distant second and third, respectively. Similar to the panel study, the 20-inch spar spacing wing box design which incorporated the honeycomb core sandwich panels resulted in the least-weight design. Unit box weights ranging from 8.3 lb/sq.ft. to 8.9 lb/sq.ft. are noted for the 20-inch and 40-inch spar spacing designs, respectively. The corresponding box weights for the truss-core sandwich range from 8.6 lb/sq.ft. to 9.9 lb/sq.ft. for the same spar spacings.

Based on the panel and wing box study results, the honeycomb core sandwich concept was selected for further valuation in the following Detail Concept Analysis.

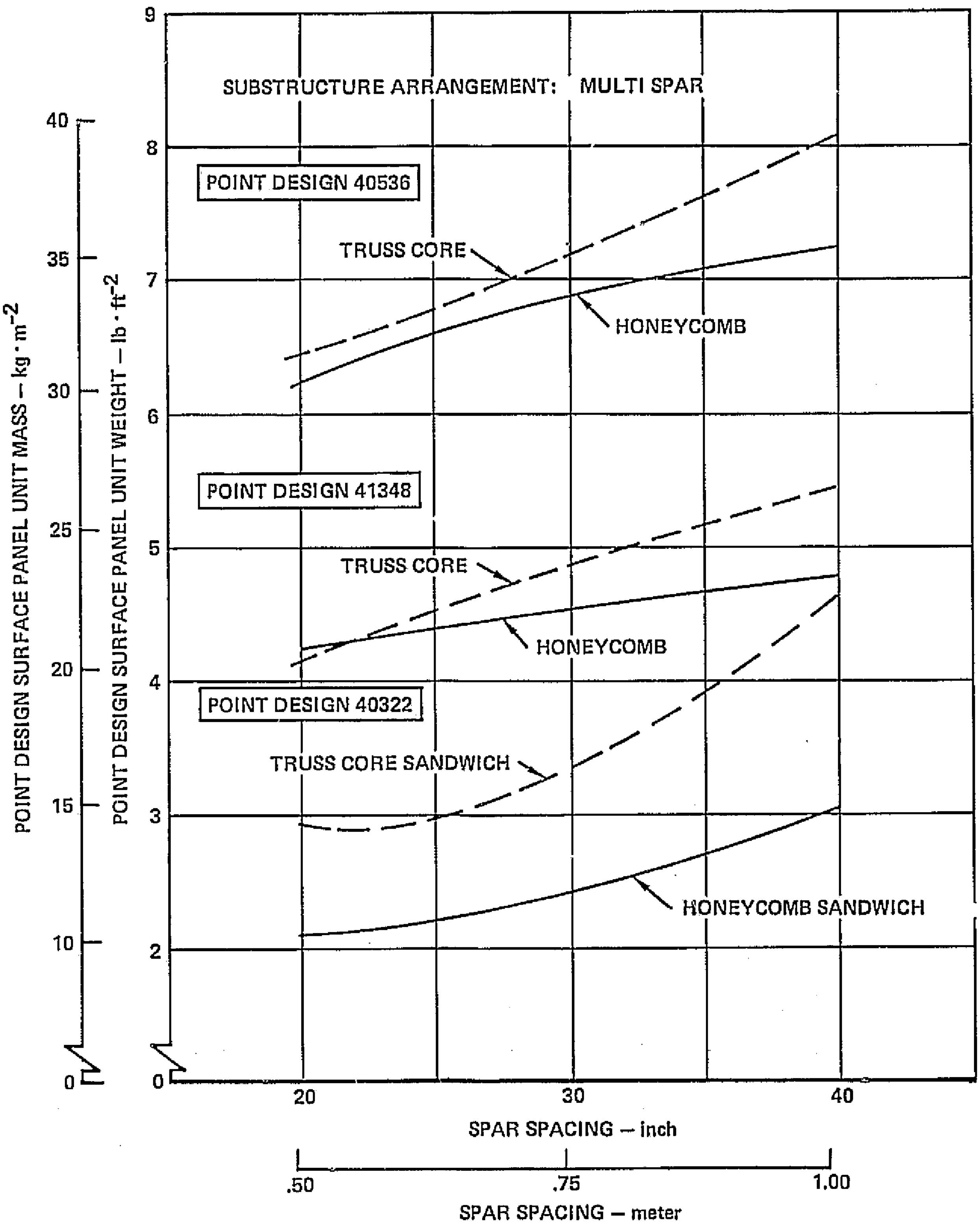


Figure 12-33. Weight Comparison of the Candidate Surface Panel Concepts for Multispar Arrangement

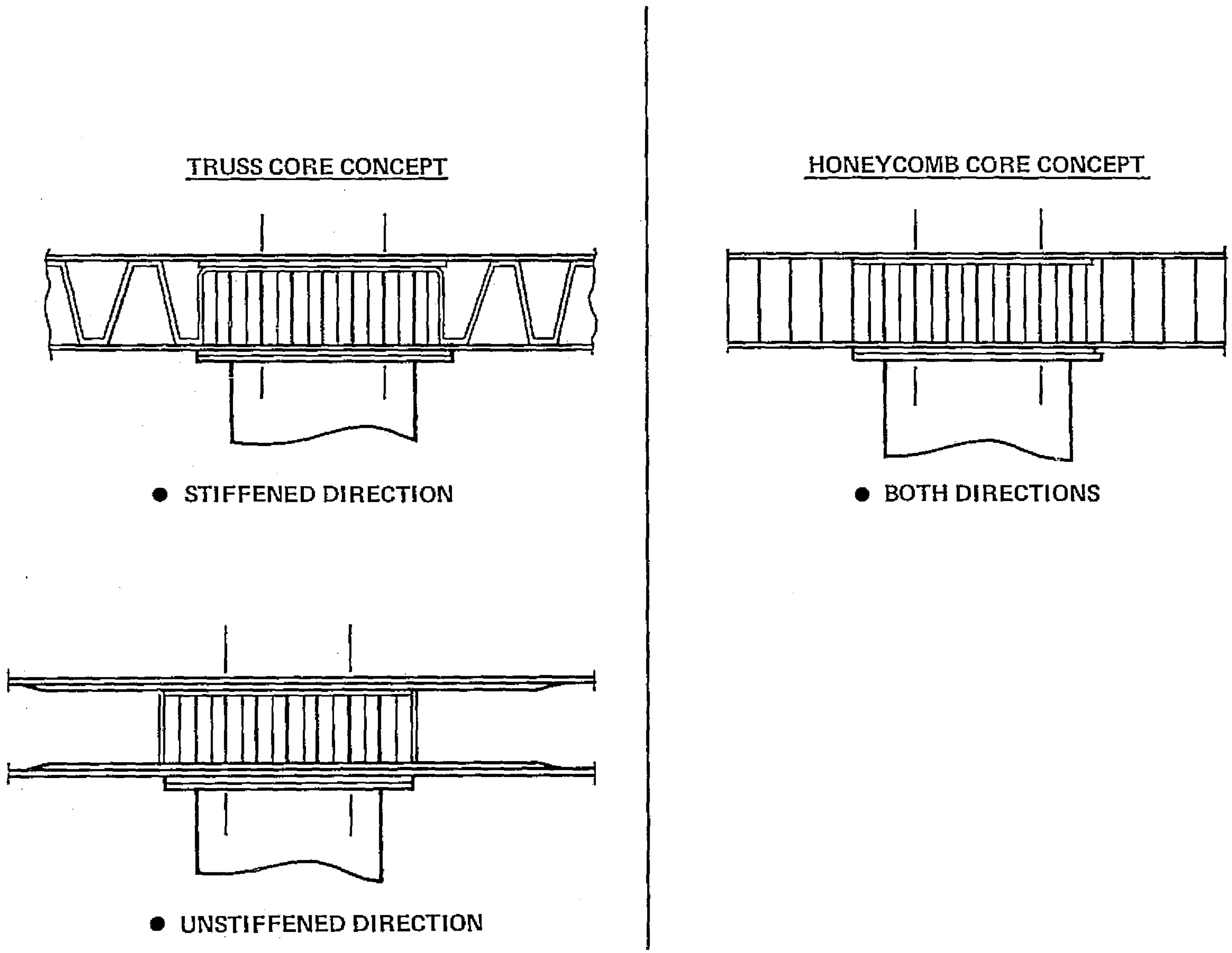


Figure 12-34. Panel Joining Methods for the Candidate Surface Panel Concepts

TABLE 12-35. DETAIL WING WEIGHT COMPARISON OF THE MONOCOQUE PANEL CONCEPT - INITIAL SCREENING

PANEL CONCEPT	HONEYCOMB CORE SANDWICH			TRUSS-CORE SANDWICH		
POINT DESIGN	40536					
SPAR SPACING (IN)	20	30	40	20	30	40
PANELS UPPER LOWER Σ	 2.92 3.30 (6.22)	 3.49 3.38 (6.87)	 3.77 3.46 (7.23)	 3.16 3.31 (6.47)	 3.73 3.47 (7.20)	 4.46 3.60 (8.06)
RIB WEBS BULKHEAD TRUSS Σ	 0.24 0.23 (0.47)	 0.24 0.23 (0.47)	 0.24 0.23 (0.47)	 0.24 0.23 (0.47)	 0.24 0.23 (0.47)	 0.24 0.23 (0.47)
SPAR WEBS BULKHEAD TRUSS Σ	 0.24 0.59 (0.83)	 0.28 0.39 (0.67)	 0.32 0.33 (0.65)	 0.24 0.59 (0.83)	 0.28 0.39 (0.67)	 0.32 0.33 (0.65)
RIB CAPS UPPER LOWER Σ	 0.07 0.07 (0.14)	 0.08 0.07 (0.15)	 0.09 0.08 (0.17)	 0.13 0.12 (0.25)	 0.15 0.10 (0.25)	 0.18 0.11 (0.29)
SPAR CAPS UPPER LOWER Σ	 0.24 0.24 (0.48)	 0.18 0.16 (0.34)	 0.14 0.14 (0.28)	 0.21 0.20 (0.41)	 0.15 0.15 (0.30)	 0.16 0.12 (0.28)
FASTENERS	0.05 (0.05)	0.04 (0.04)	0.03 (0.03)	0.05 (0.05)	0.04 (0.04)	0.03 (0.03)
WEB INTERSECTION	0.13 (0.13)	0.12 (0.12)	0.11 (0.11)	0.13 (0.13)	0.12 (0.12)	0.11 (0.11)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	8.32	8.66	8.94	8.61	9.05	9.89

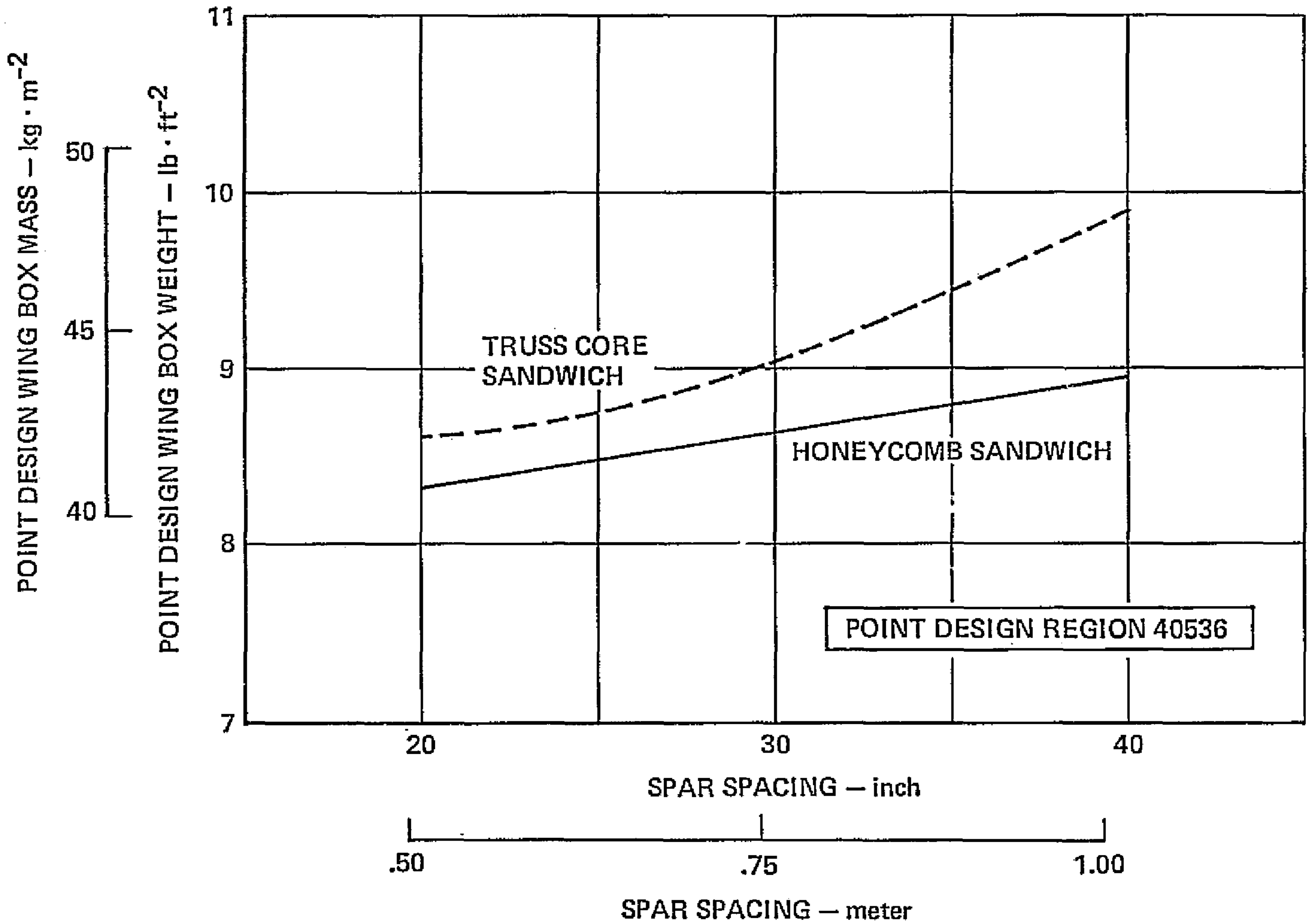


Figure 12-35. Wing Box Weight Comparison of the Candidate - Monocoque Surface Panel Concepts - Initial Screening

Monocoque Detail Concept Analysis

The most promising panel concept surviving the Initial Screening analysis was the honeycomb-core sandwich panel concept with the general panel proportion commensurate with a multispar structural arrangement. In addition, only one material was considered for this investigation, titanium alloy Ti-6Al-4V (ANN).

For this analysis, representative wing box segments were subjected to a weight evaluation at the six study point design regions. These wing box segments included the wing panels, representative substructure, and the related non-optimum factors. In addition, unit wing box weights were calculated to reflect the specific method of attaching the panels to the substructure.

Panel Analysis - In addition to the panel proportions defined for the initial screening analysis, the honeycomb sandwich panels were sized for three additional point design regions: 40236, 41036, and 41316. These panels were analyzed for their most critical load/temperature environments (Reference Table 12-27) and the results are summarized in Table 12-36. This table includes the specific panel cross sectional dimensions and related mass data for each of the new point design regions.

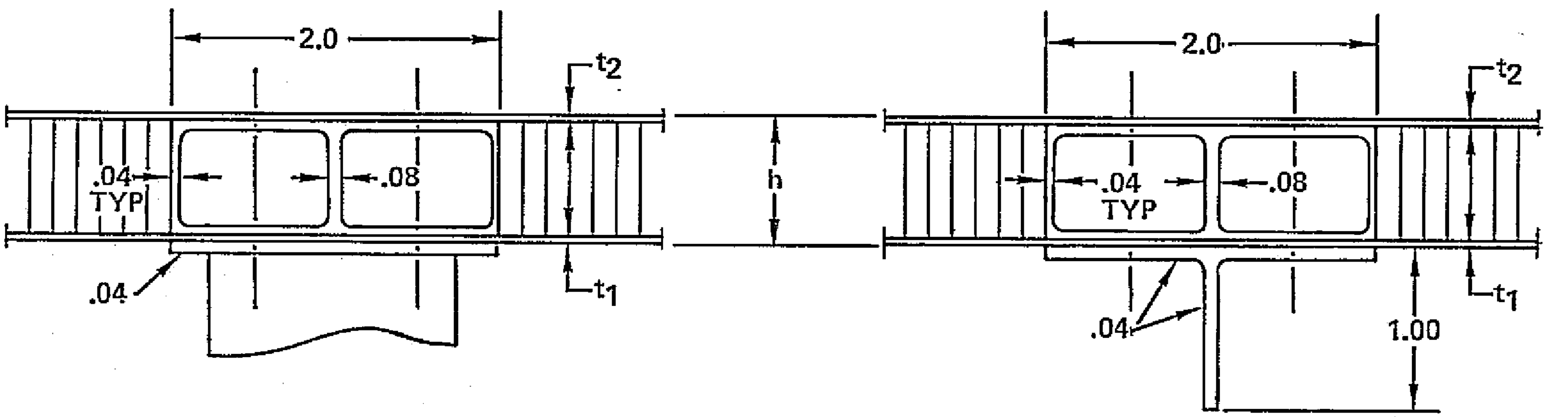
Substructure Analysis - Typical substructure was investigated for application to the monocoque structural arrangement. This substructure included the following components: spar caps and webs, rib caps and webs, and the applicable non-optimum factors. The weight of the rib and spar caps varied with the specific type of panel-to-substructure attachment being considered; whereas, the remaining substructure components (rib and spar webs) were invariant with the attachment design.

The three types of panel-to-substructure junctions considered in this analysis are shown in Figures 12-36, 12-37, and 12-38. The first type (Figure 12-36) consists of embedding tubular inserts into the honeycomb panel at the rib and spar intersections and mechanically fastening the structural components. Figure 12-37 presents the second type, which also uses a tubular insert that is welded into the panel, and to the rib and spar attachments. The last type of panel-to-substructure junction considered is shown in Figure 12-38 and is comprised of a densified core insert which is mechanically fastened to the substructure.

TABLE 12-36. PANEL GEOMETRY AND WEIGHT FOR THE MULTISPAR DESIGN HONEYCOMB SANDWICH PANELS

POINT DESIGN REGION		40236						41036						41316					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
RIB SPACING	(m)	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52
	(in.)	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60
ASPECT RATIO		0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67
DIMENSIONS:																			
H	(cm)	1.958	2.883	3.818	1.059	1.412	2.360	1.760	2.525	3.254	0.655	0.848	1.095	1.834	2.545	3.409	0.488	0.622	0.808
	(in.)	0.771	1.135	1.503	0.417	0.556	0.926	0.693	0.994	1.281	0.258	0.334	0.431	0.722	1.002	1.342	0.192	0.245	0.318
t_1	(cm)	0.119	0.122	0.130	0.234	0.188	0.147	0.084	0.084	0.091	0.119	0.127	0.114	0.132	0.137	0.145	0.178	0.173	0.173
	(in.)	0.047	0.048	0.051	0.092	0.074	0.058	0.033	0.033	0.036	0.047	0.050	0.045	0.052	0.054	0.057	0.070	0.068	0.068
t_2	(cm)	0.124	0.127	0.130	0.069	0.117	0.157	0.084	0.094	0.094	0.091	0.081	0.097	0.135	0.137	0.135	0.157	0.163	0.163
	(in.)	0.049	0.050	0.051	0.027	0.046	0.062	0.033	0.037	0.037	0.036	0.032	0.038	0.053	0.054	0.053	0.062	0.064	0.064
t_c	(cm)	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005
	(in.)	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002
s	(cm)	0.947	0.744	0.729	1.270	1.270	1.270	0.747	0.767	0.752	1.270	1.270	1.270	0.925	0.831	0.726	1.270	1.270	1.270
	(in.)	0.373	0.293	0.287	0.500	0.500	0.500	0.294	0.302	0.296	0.500	0.500	0.500	0.364	0.327	0.286	0.500	0.500	0.500
MASS DATA:																			
$\bar{t}$	(cm)	0.264	0.284	0.310	0.310	0.315	0.323	0.191	0.208	0.226	0.213	0.213	0.218	0.284	0.302	0.323	0.335	0.335	0.338
	(in.)	0.104	0.112	0.122	0.122	0.124	0.127	0.075	0.082	0.089	0.084	0.084	0.086	0.112	0.119	0.127	0.132	0.132	0.133
W	($kg \cdot m^{-2}$)	11.674	12.631	13.715	13.720	13.964	14.305	8.393	9.228	9.989	9.443	9.477	9.623	12.597	13.412	14.252	14.857	14.891	15.009
	($lb \cdot ft^{-2}$)	2.391	2.587	2.809	2.810	2.860	2.930	1.719	1.890	2.046	1.934	1.941	1.971	2.580	2.747	2.919	3.043	3.050	3.074
W_c	($kg \cdot m^{-2}$)	0.815	1.592	2.192	0.269	0.391	0.723	0.962	1.377	1.836	0.156	0.225	0.312	0.762	1.230	1.938	0.054	0.103	0.166
	($lb \cdot ft^{-2}$)	0.167	0.326	0.449	0.055	0.080	0.148	0.197	0.282	0.376	0.032	0.046	0.064	0.156	0.252	0.397	0.011	0.021	0.034
ρ_c	($kg \cdot m^{-3}$)	47.687	60.390	61.655	35.433	35.433	35.433	60.277	58.724	59.861	35.433	35.433	35.433	48.616	54.174	61.959	35.433	35.433	35.433
	($lb \cdot ft^{-3}$)	2.977	3.770	3.849	2.212	2.212	2.212	3.763	3.666	3.737	2.212	2.212	2.212	3.035	3.382	3.868	2.212	2.212	2.212
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31

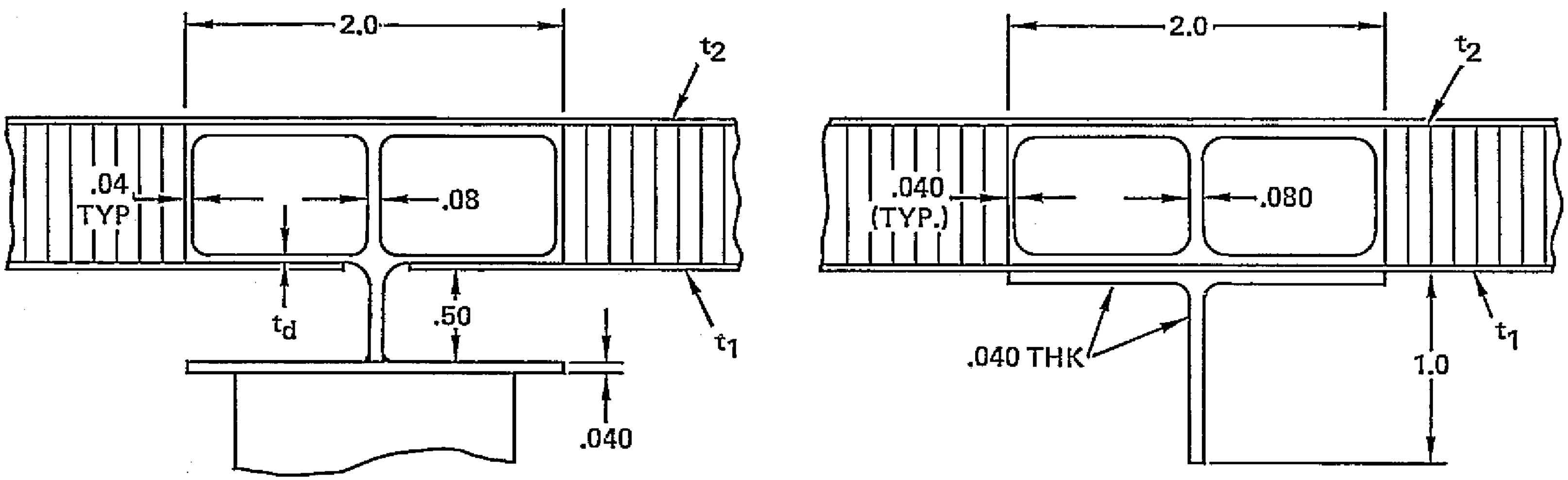
NOTE: (1) ASPECT RATIO = $L_{P,X}/L_{P,Y}$
(2) BRAZE MATERIAL NOT INCLUDED



• BULKHEAD JUNCTURE

• INTERMEDIATE TRUSS JUNCTURE

Figure 12-36. Mechanically Fastened - Tubular Insert
Panel-to-Substructure Junctures



• BULKHEAD JUNCTURE

• INTERMEDIATE TRUSS JUNCTURE

Figure 12-37. Welded - Tubular Insert Panel-to-Substructure Junctures

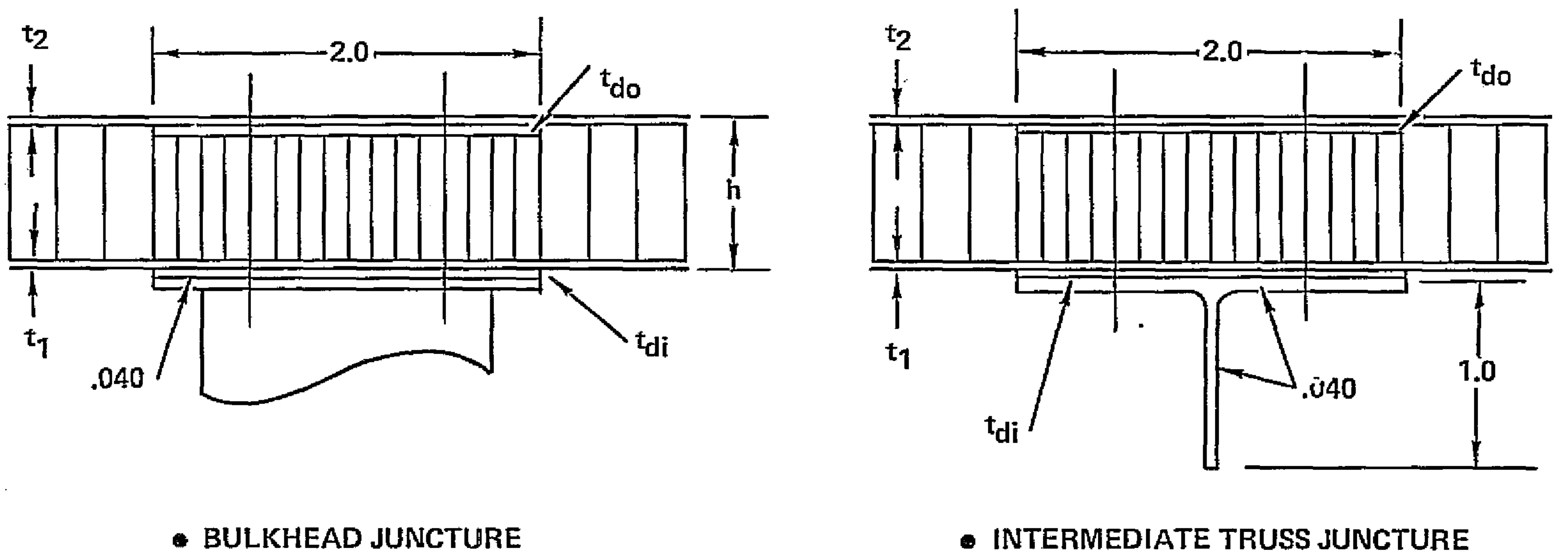


Figure 12-38. Mechanically Fastened - Densified Core Panel-to-Substructure Junctures

For the vertical web designs, combinations of circular-arc and truss-type webs were used as dictated by the specific design requirements at each point design region. For example, at region 40536 which is located in a fuel tank, circular-arc webs were used for the fuel tank bulkheads and truss webs for the intermediate spars and ribs.

The weights of the substructure components are itemized in the detail weight statements for each of the wing structural arrangements.

Monocoque Box Weights - A detail weight statement and the optimum rib/spar spacings were determined for each of the monocoque wing arrangements. These arrangements, as characterized by the type of panel-to-substructure junction design, are all multi-spar arrangements and employ the honeycomb-core sandwich panel concept.

Mechanically Fastened-Tubular Insert - A detail weight statement for this arrangement is shown in Table 12-37. This data reflects the weight/strength analysis conducted at point design regions 40322, 40536 and 41348. In addition, this data includes a variable spar spacing of 20-inches, 30-inches, and 40-inches with a constant rib spacing.

In addition to the detail weight tables these weights are presented graphically in Figures 12-39, 12-40, and 12-41, for regions 40322, 40536, and 41348, respectively. The forward wing box region 40322, displayed in Figure 12-39, has an optimum design for a spar spacing between 25-inches and 30-inches with a total wing box weight of approximately 4.4 lb/sq.ft. For region 40536 (Figure 12-40) no discernable optimum spar spacing is indicated for the positive sloping total weight curve. The least-weight design is for 20-inch spar spacing, and weighs approximately 8.7 lb/sq.ft. The total weight curve for region 41348 is shown in Figure 12-41. A minimum weight of 5.8 lb/sq.ft. occurs for the smallest spar spacing investigated, 20-inches.

Welded-Tubular Insert - Detail weight statements are shown in Tables 12-38 and 12-39 for the six point design regions investigated. This data reflects a multispar arrangement with a constant rib spacing and variable spar spacings of 20-inches, 30-inches, and 40-inches. For ease in interpretation, this weight data is shown graphically in Figures 12-42 through 12-47. No discernable optimum design is noted for any of the regions with the exception of region 40322. The total weight

TABLE 12-37. DETAIL WING WEIGHTS FOR THE MONOCOQUE MECHANICALLY FASTENED - TUBULAR INSERT ARRANGEMENT

POINT DESIGN REGION	40322			40536			41348		
SPAR SPAC (IN.)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	1.144	1.308	1.608	2.918	3.489	3.772	2.018	2.310	2.469
LOWER	0.980	1.126	1.445	3.302	3.377	3.458	2.241	2.224	2.288
$\sum$	(2.124)	(2.434)	(3.053)	(6.220)	(6.866)	(7.230)	(4.259)	(4.534)	(4.757)
RIB WEBS									
BULKHEAD	0.241	0.241	0.241	0.244	0.244	0.244	0.100	0.100	0.100
TRUSS	0.198	0.198	0.198	0.229	0.229	0.229	–	–	–
$\sum$	(0.439)	(0.439)	(0.439)	(0.473)	(0.473)	(0.473)	(0.100)	(0.100)	(0.100)
SPAR WEBS									
BULKHEAD	0.355	0.343	0.352	0.245	0.285	0.321	0.401	0.301	0.251
TRUSS	0.339	0.194	0.121	0.590	0.389	0.326	–	–	–
$\sum$	(0.694)	(0.537)	(0.473)	(0.835)	(0.674)	(0.647)	(0.401)	(0.301)	(0.251)
RIB CAPS									
UPPER	0.120	0.140	0.158	0.127	0.143	0.150	0.127	0.146	0.154
LOWER	0.133	0.138	0.193	0.100	0.106	0.127	0.099	0.106	0.113
$\sum$	(0.253)	(0.278)	(0.351)	(0.227)	(0.249)	(0.277)	(0.226)	(0.252)	(0.267)
SPAR CAPS									
UPPER	0.401	0.307	0.268	0.418	0.311	0.242	0.376	0.289	0.227
LOWER	0.445	0.316	0.325	0.332	0.236	0.206	0.290	0.208	0.167
$\sum$	(0.846)	(0.623)	(0.591)	(0.750)	(0.547)	(0.448)	(0.666)	(0.497)	(0.394)
NON-OPTIMUM									
MECH. FAST.	0.050	0.040	0.030	0.050	0.040	0.030	0.050	0.040	0.030
WEB INTERS.	0.113	0.098	0.091	0.131	0.115	0.112	0.050	0.040	0.035
$\sum$	(0.163)	(0.138)	(0.121)	(0.181)	(0.155)	(0.142)	(0.100)	(0.080)	(0.065)
$\sum$ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	4.519	4.449	5.028	8.686	8.964	9.217	5.752	5.764	5.834

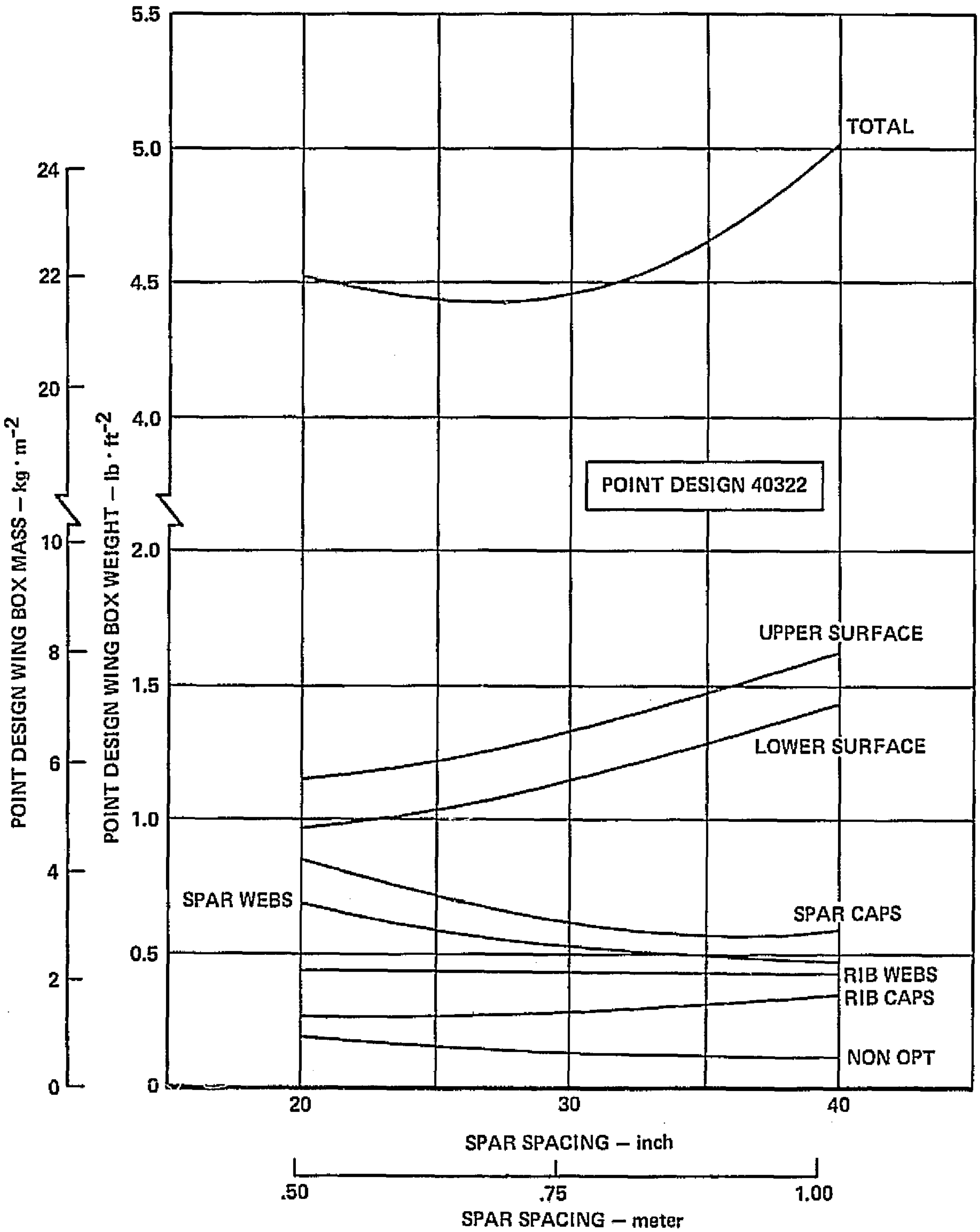


Figure 12-39. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Tubular Insert Arrangement, Point Design Region 40322

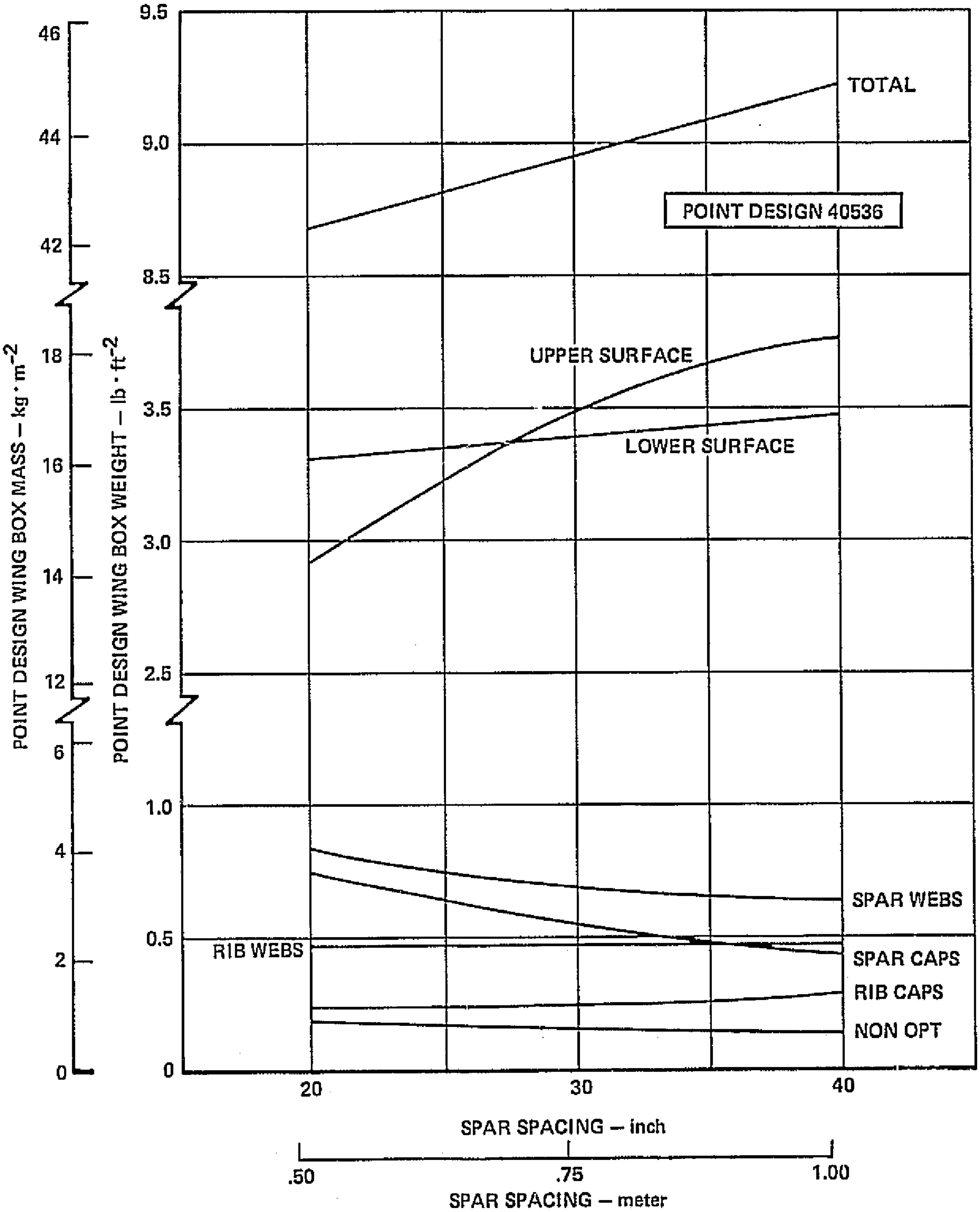


Figure 12-40. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Tubular Insert Arrangement, Point Design Region 40536

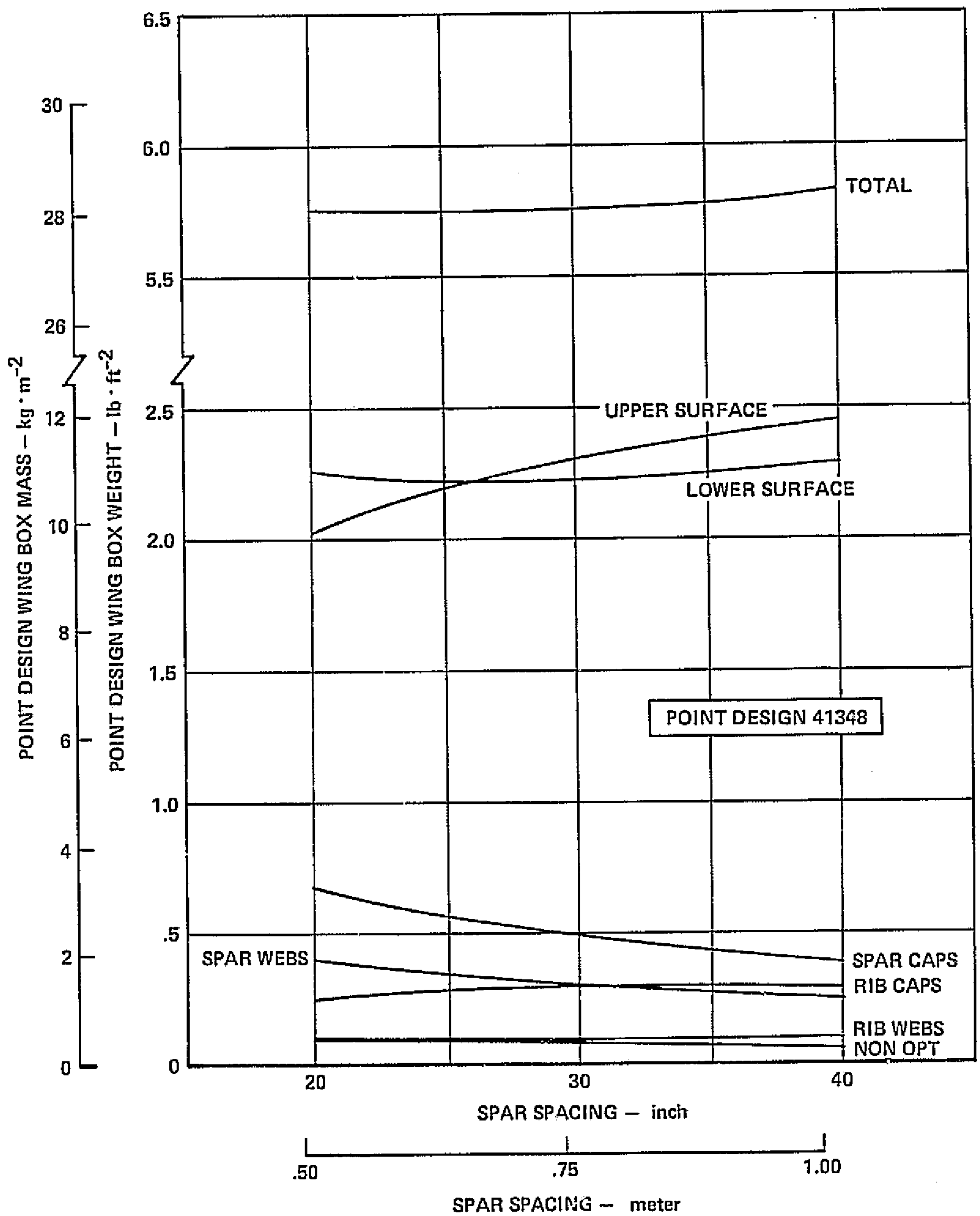


Figure 12-41. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Tubular Insert Arrangement, Point Design Region 41348

TABLE 12-38. DETAIL WING WEIGHTS FOR THE MONOCOQUE WELDED - TUBULAR INSERT ARRANGEMENT

POINT DESIGN REGION	40322			41316			41348		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	1.144	1.308	1.608	2.710	2.957	3.189	2.018	2.310	2.469
LOWER	0.980	1.126	1.445	3.143	3.150	3.174	2.241	2.224	2.288
Σ	(2.124)	(2.434)	(3.053)	(5.853)	(6.107)	(6.363)	(4.259)	(4.534)	(4.757)
RIB WEBS									
BULKHEAD	0.241	0.241	0.241	0.187	0.187	0.187	0.100	0.100	0.100
TRUSS	0.198	0.198	0.198	—	—	—	—	—	—
Σ	(0.439)	(0.439)	(0.439)	(0.187)	(0.187)	(0.187)	(0.100)	(0.100)	(0.100)
SPAR WEBS									
BULKHEAD	0.355	0.343	0.352	0.289	0.229	0.190	0.401	0.301	0.251
TRUSS	0.339	0.194	0.121	—	—	—	—	—	—
Σ	(0.694)	(0.537)	(0.473)	(0.289)	(0.229)	(0.190)	(0.401)	(0.301)	(0.251)
RIB CAPS									
UPPER	0.114	0.134	0.158	0.139	0.154	0.173	0.141	0.160	0.168
LOWER	0.127	0.133	0.188	0.090	0.129	0.117	0.113	0.123	0.127
Σ	(0.241)	(0.267)	(0.341)	(0.229)	(0.283)	(0.290)	(0.254)	(0.283)	(0.295)
SPAR CAPS									
UPPER	0.355	0.285	0.258	0.463	0.341	0.258	0.419	0.317	0.248
LOWER	0.399	0.294	0.310	0.304	0.288	0.177	0.332	0.241	0.188
Σ	(0.754)	(0.579)	(0.568)	(0.767)	(0.629)	(0.435)	(0.751)	(0.558)	(0.436)
NON-OPTIMUM									
MECH. FAST.	—	—	—	—	—	—	—	—	—
WEB INTERS.	0.113	0.098	0.091	0.048	0.042	0.035	0.050	0.040	0.035
Σ	(0.113)	(0.098)	(0.091)	(0.048)	(0.042)	(0.035)	(0.050)	(0.040)	(0.035)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	4.365	4.364	4.965	7.373	7.477	7.500	5.815	5.816	5.874

TABLE 12-39. DETAIL WING WEIGHTS FOR THE MONOCOQUE WELDED - TUBULAR INSERT ARRANGEMENT

POINT DESIGN REGION	40236			40536			41036		
Σ SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS									
UPPER	2.511	2.817	3.109	2.918	3.489	3.772	1.869	2.090	2.246
LOWER	2.910	2.960	3.030	3.302	3.377	3.458	1.944	2.041	2.071
Σ	(5.421)	(5.777)	(6.139)	(6.220)	(6.866)	(7.230)	(3.813)	(4.131)	(4.317)
RIB WEBS									
BULKHEAD	0.329	0.329	0.329	0.244	0.244	0.244	0.126	0.126	0.126
TRUSS	0.396	0.396	0.396	0.229	0.229	0.229	0.111	0.111	0.111
Σ	(0.725)	(0.725)	(0.725)	(0.473)	(0.473)	(0.473)	(0.237)	(0.237)	(0.237)
SPAR WEBS									
BULKHEAD	0.367	0.422	0.463	0.245	0.285	0.321	0.096	0.114	0.130
TRUSS	0.877	0.706	0.514	0.590	0.389	0.326	0.188	0.183	0.165
Σ	(1.244)	(1.128)	(0.977)	(0.835)	(0.674)	(0.647)	(0.284)	(0.297)	(0.295)
RIB CAPS									
UPPER	0.127	0.145	0.157	0.123	0.139	0.146	0.124	0.138	0.139
LOWER	0.105	0.115	0.136	0.096	0.102	0.123	0.099	0.104	0.125
Σ	(0.232)	(0.260)	(0.293)	(0.219)	(0.241)	(0.269)	(0.223)	(0.242)	(0.264)
SPAR CAPS									
UPPER	0.378	0.273	0.247	0.376	0.292	0.235	0.367	0.278	0.188
LOWER	0.313	0.234	0.209	0.290	0.217	0.199	0.288	0.206	0.169
Σ	(0.691)	(0.507)	(0.456)	(0.666)	(0.509)	(0.434)	(0.655)	(0.484)	(0.357)
NON-OPTIMUM									
MECH. FAST.	—	—	—	—	—	—	—	—	—
WEB INTERS.	0.197	0.185	0.170	0.131	0.115	0.112	0.052	0.053	0.053
Σ	(0.197)	(0.185)	(0.170)	(0.131)	(0.115)	(0.112)	(0.052)	(0.053)	(0.053)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	8.510	8.582	8.760	8.544	8.878	9.165	5.264	5.444	5.523

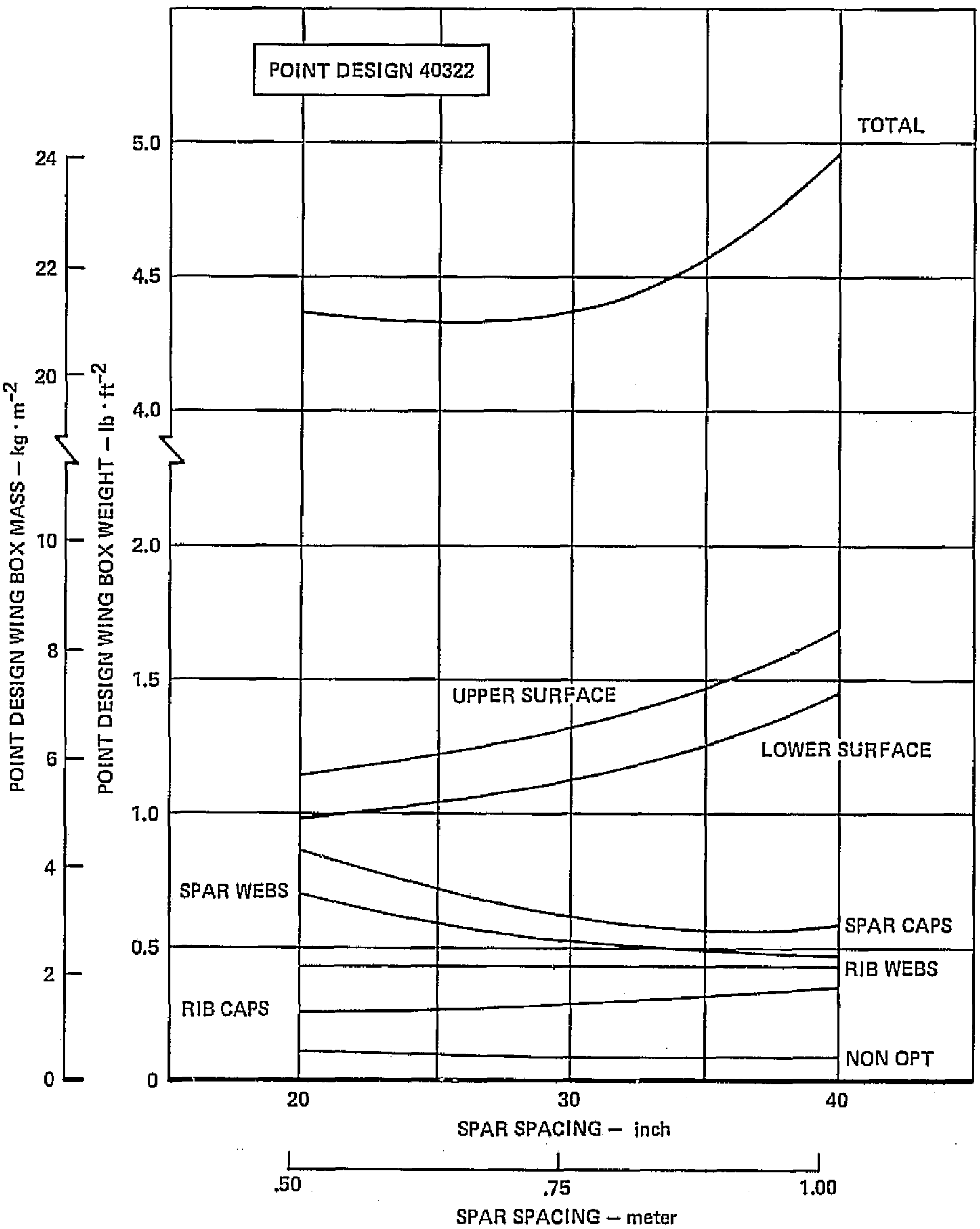


Figure 12-42. Optimum Spar Spacing for the Monocoque Welded - Tubular Insert Arrangement, Point Design Region 40322

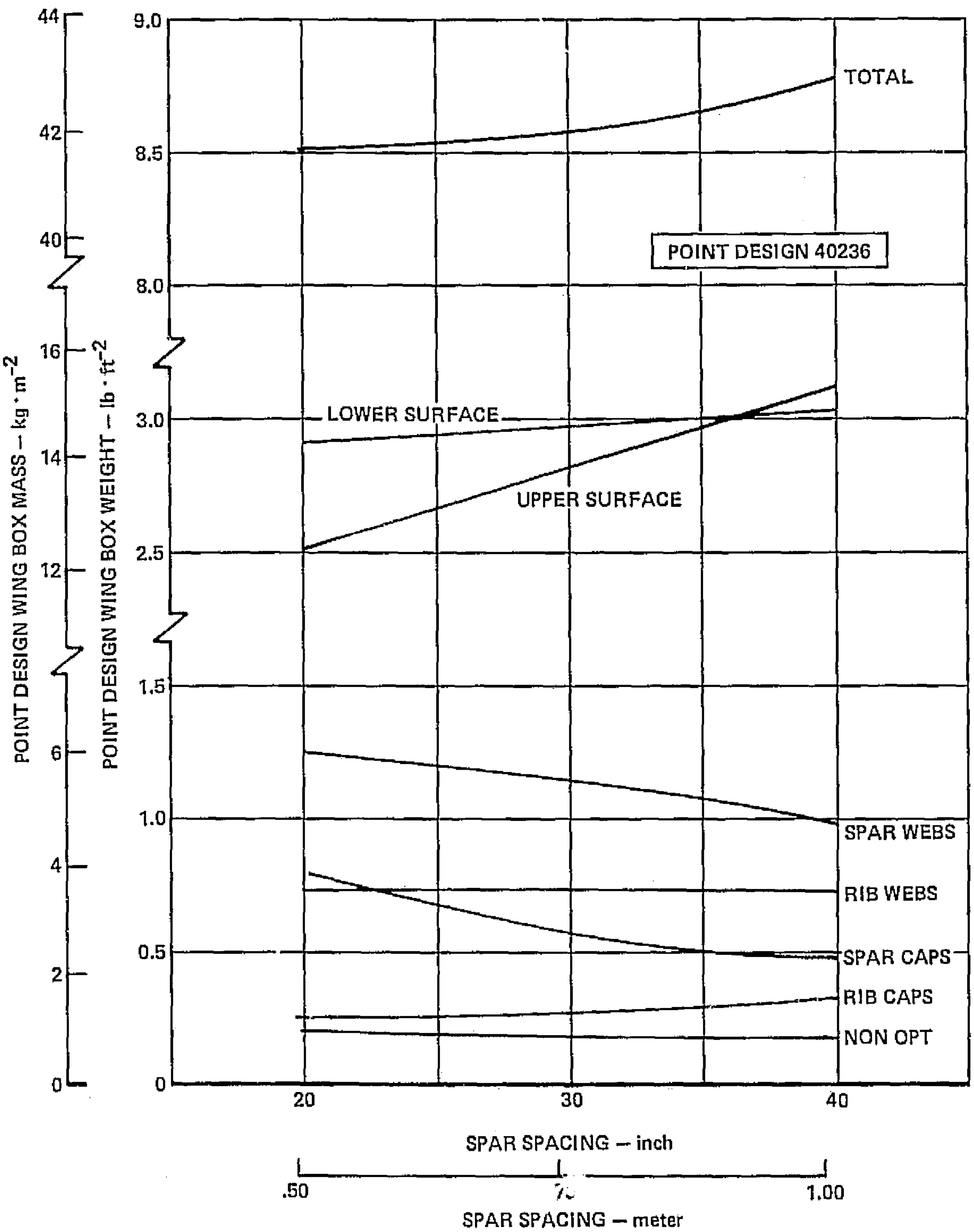


Figure 12-43. Optimum Spar Spacing for the Monocoque Welded - Tubular Insert Arrangement, Point Design Region 40236

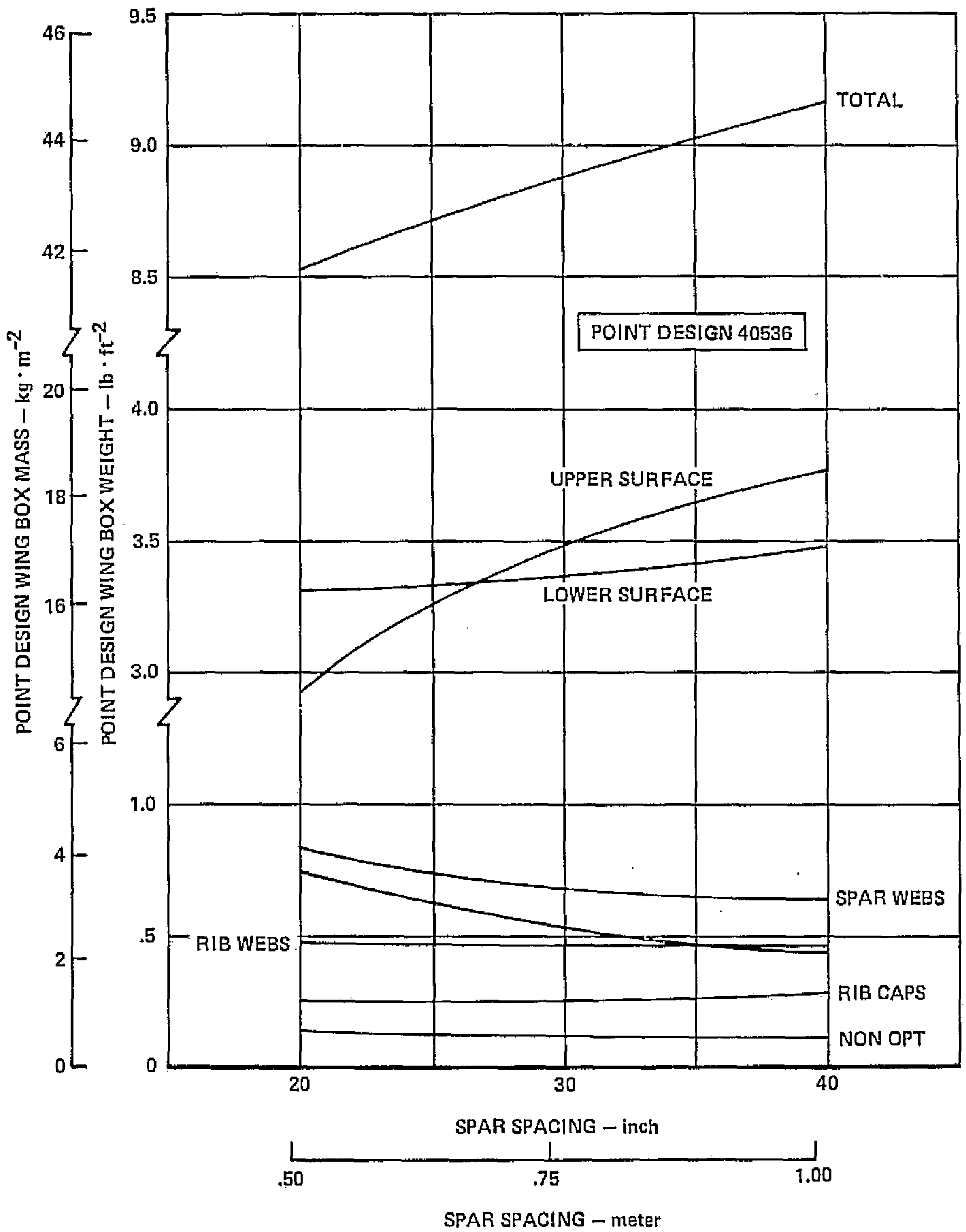


Figure 12-44. Optimum Spar Spacing for the Monocoque Welded - Tubular Insert Arrangement, Point Design Region 40536

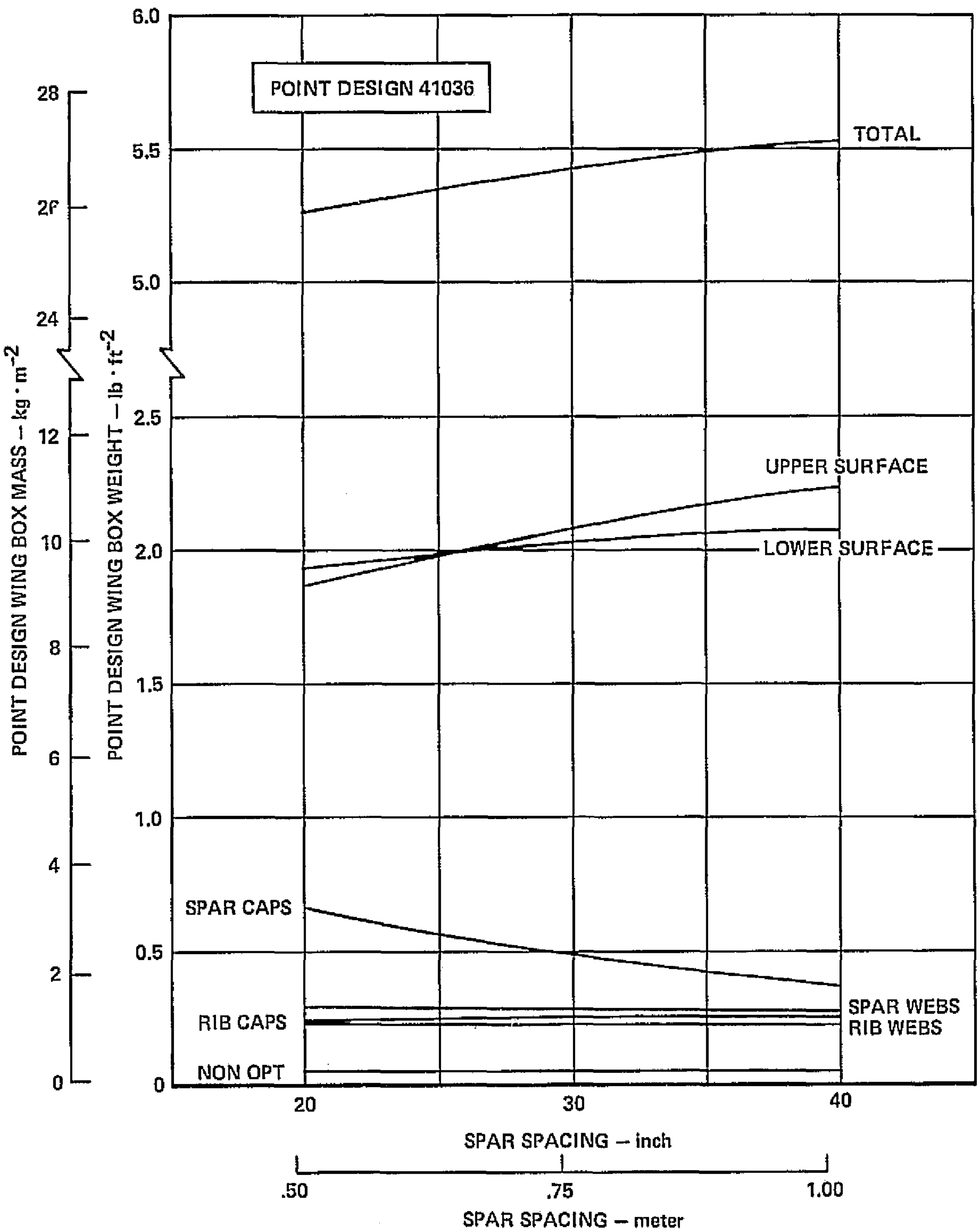


Figure 12-45. Optimum Spar Spacing for the Monocoque Welded - Tubular Insert Arrangement, Point Design Region 41036

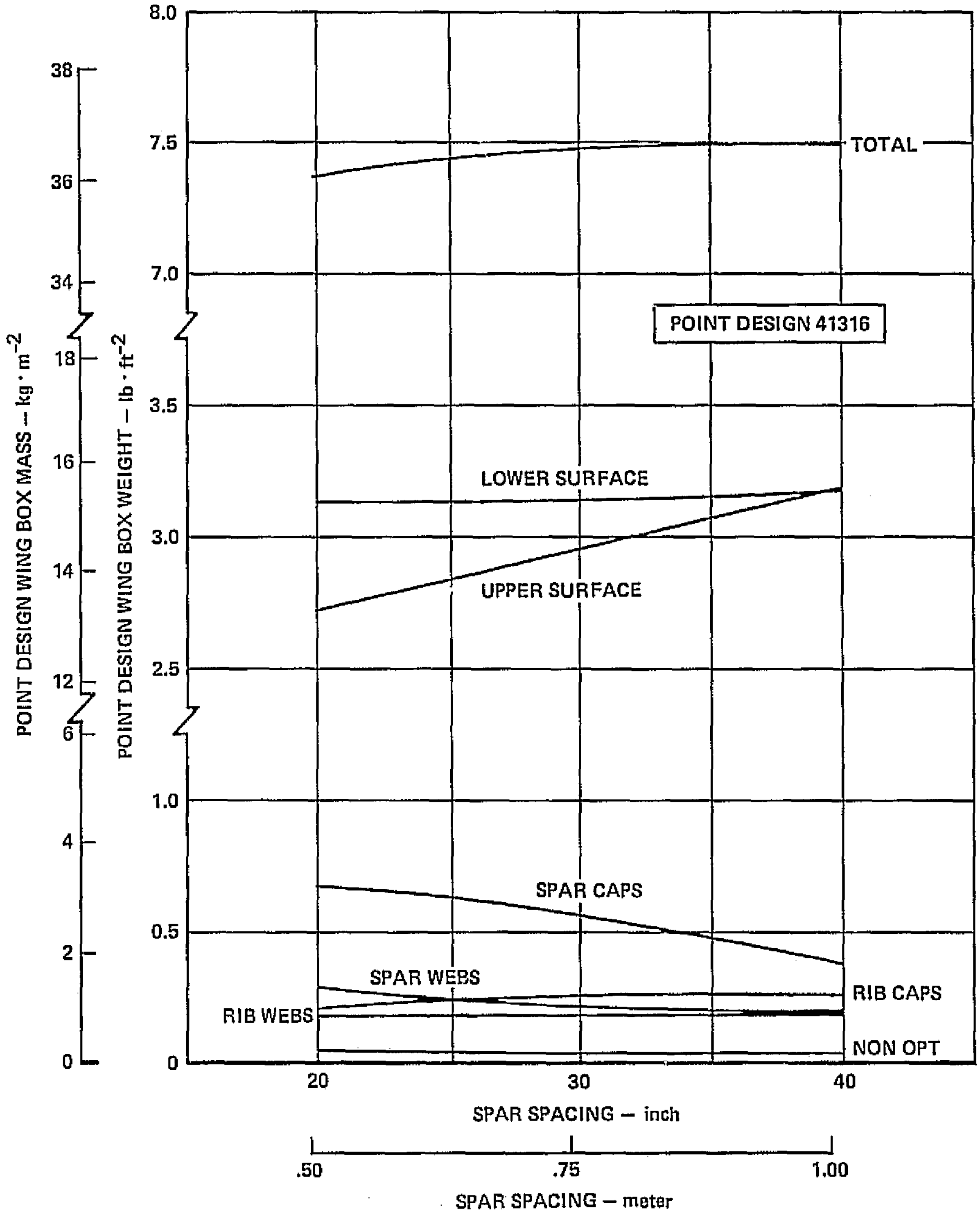


Figure 12-46. Optimum Spar Spacing for the Monocoque Welded - Tubular Insert Arrangement, Point Design Region 41316

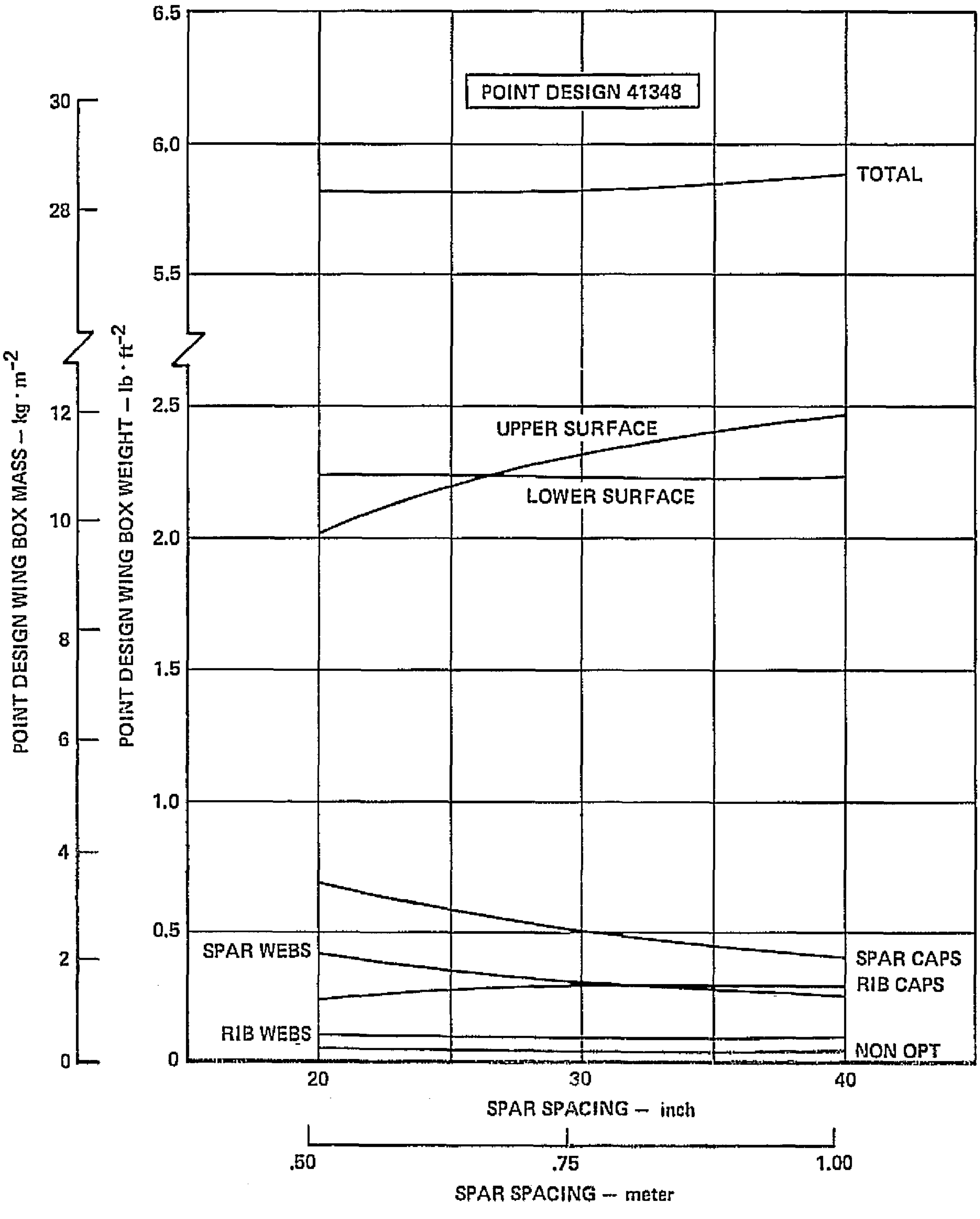


Figure 12-47. Optimum Spar Spacing for the Monocoque Welded - Tubular Insert Arrangement, Point Design Region 41348

curve for this region is shown in Figure 12-42 and indicate a minimum-weight design occurring for a spar spacing between 20- and 30-inches.

Mechanically Fastened-Densified Core - The detail weight statements for this arrangement are presented in Tables 12-40 through 12-41. Similar to the other arrangements, the weight data for this arrangement are also shown graphically in Figures 12-48 through 12-53. No readily discernable optimum designs are noted, but the total weights curves for regions 40322, 40236, 41316, and 41348 indicate the likelihood of the lowest spar spacing design (20-inch) investigated being an inflection point.

For summary purposes, the unit wing box weights for each of the candidate wing arrangements are presented in Table 12-42. This data represents the unit weights for the 20-inch spar spacing designs normalized to the weight of the least-weight arrangement. With reference to this table, the mechanically fastened-densified core arrangement is the minimum-weight monocoque arrangement with the welded-tubular and mechanically fastened-tubular arrangements ranked second and third, respectively. Comparing the mechanically fastened-densified core arrangement (least-weight) to the welded-tubular arrangement, a minimum weight savings of 3-percent is noted for regions 40236 and 40536; while region 40322 affords a maximum weight saving of 9-percent. Similarly, the least-weight arrangement indicates minimum and maximum weight savings of 4-percent and 13-percent, respectively, as compared to the heaviest-weight mechanically fastened - tubular arrangement.

In conclusions, based on the results of the relatively extensive Task I analysis the mechanically fastened-densified core arrangement was selected as the most-promising monocoque arrangement.

COMPOSITE REINFORCED WING ARRANGEMENT - TASK I

The Task I analysis included a relatively comprehensive study on the application of composite to the arrow-wing primary structure. This study was based on near-term technology and limited the use of composite to reinforcing the titanium primary structure on the chordwise wing arrangement. Those structural components presenting

TABLE 12-40. DETAIL WING WEIGHTS FOR THE MONOCOQUE MECHANICALLY FASTENED - DENSIFIED CORE ARRANGEMENT

POINT DESIGN REGION	40322			41316			41348		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS UPPER LOWER Σ	1.144 0.980 (2.124)	1.308 1.126 (2.434)	1.608 1.445 (3.053)	2.710 3.143 (5.853)	2.957 3.150 (6.107)	3.189 3.174 (6.363)	2.018 2.241 (4.259)	2.310 2.224 (4.534)	2.469 2.288 (4.757)
RIB WEBS BULKHEAD TRUSS Σ	0.241 0.198 (0.439)	0.241 0.198 (0.439)	0.241 0.198 (0.439)	0.187 — (0.187)	0.187 — (0.187)	0.187 — (0.187)	0.100 — (0.100)	0.100 — (0.100)	0.100 — (0.100)
SPAR WEBS BULKHEAD TRUSS Σ	0.355 0.339 (0.694)	0.343 0.194 (0.537)	0.352 0.121 (0.473)	0.289 — (0.289)	0.229 — (0.229)	0.190 — (0.190)	0.401 — (0.401)	0.301 — (0.301)	0.251 — (0.251)
RIB CAPS UPPER LOWER Σ	0.069 0.062 (0.131)	0.074 0.072 (0.146)	0.068 0.066 (0.134)	0.070 0.067 (0.137)	0.077 0.067 (0.144)	0.077 0.067 (0.144)	0.077 0.082 (0.159)	0.079 0.087 (0.166)	0.084 0.103 (0.187)
SPAR CAPS UPPER LOWER Σ	0.235 0.214 (0.449)	0.163 0.168 (0.331)	0.127 0.113 (0.240)	0.232 0.224 (0.456)	0.165 0.149 (0.314)	0.115 0.101 (0.216)	0.227 0.240 (0.467)	0.156 0.170 (0.326)	0.124 0.153 (0.277)
NON-OPTIMUM MECH. FAST. WEB INTERS. Σ	0.050 0.113 (0.163)	0.040 0.098 (0.138)	0.03 0.091 (0.121)	0.050 0.048 (0.098)	0.040 0.042 (0.082)	0.030 0.030 (0.060)	0.05 0.050 (0.10)	0.04 0.04 (0.08)	0.03 0.035 (0.065)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	4.00	4.025	4.46	7.02	7.063	7.16	5.486	5.507	5.637

TABLE 12-41. DETAIL WING WEIGHTS FOR THE MONOCOQUE MECHANICALLY FASTENED - DENSIFIED CORE ARRANGEMENT

POINT DESIGN REGION	40236			40536			41036		
SPAR SPAC (IN)	20	30	40	20	30	40	20	30	40
PANELS UPPER LOWER Σ	 2.511 2.910 (5.421)	 2.817 2.960 (5.777)	 3.109 3.030 (6.139)	 2.918 3.302 (6.220)	 3.489 3.377 (6.866)	 3.772 3.458 (7.230)	 1.869 1.944 (3.813)	 2.090 2.041 (4.131)	 2.246 2.071 (4.317)
RIB WEBS BULKHEAD TRUSS Σ	 0.329 0.396 (0.725)	 0.329 0.396 (0.725)	 0.329 0.396 (0.725)	 0.244 0.229 (0.473)	 0.244 0.229 (0.473)	 0.244 0.229 (0.473)	 0.126 0.111 (0.237)	 0.126 0.111 (0.237)	 0.126 0.111 (0.237)
SPAR WEBS BULKHEAD TRUSS Σ	 0.367 0.877 (1.244)	 0.422 0.706 (1.128)	 0.463 0.514 (0.977)	 0.245 0.590 (0.835)	 0.285 0.389 (0.674)	 0.321 0.326 (0.647)	 0.096 0.188 (0.284)	 0.114 0.183 (0.297)	 0.130 0.165 (0.295)
RIB CAPS UPPER LOWER Σ	 0.073 0.073 (0.146)	 0.073 0.073 (0.146)	 0.075 0.083 (0.158)	 0.071 0.071 (0.142)	 0.083 0.073 (0.156)	 0.090 0.081 (0.171)	 0.076 0.069 (0.145)	 0.079 0.074 (0.153)	 0.080 0.075 (0.155)
SPAR CAPS UPPER LOWER Σ	 0.241 0.241 (0.482)	 0.164 0.164 (0.328)	 0.130 0.126 (0.256)	 0.239 0.239 (0.478)	 0.181 0.163 (0.344)	 0.144 0.136 (0.280)	 0.250 0.250 (0.471)	 0.169 0.169 (0.329)	 0.122 0.117 (0.239)
NON-OPTIMUM MECH. FAST. WEB INTERS. Σ	 0.050 0.197 (0.247)	 0.040 0.185 (0.225)	 0.030 0.170 (0.200)	 0.050 0.131 (0.181)	 0.04 0.115 (0.155)	 0.03 0.112 (0.142)	 0.050 0.052 (0.102)	 0.040 0.053 (0.093)	 0.030 0.054 (0.084)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	8.265	8.239	8.455	8.329	8.668	8.943	5.052	5.24	5.327

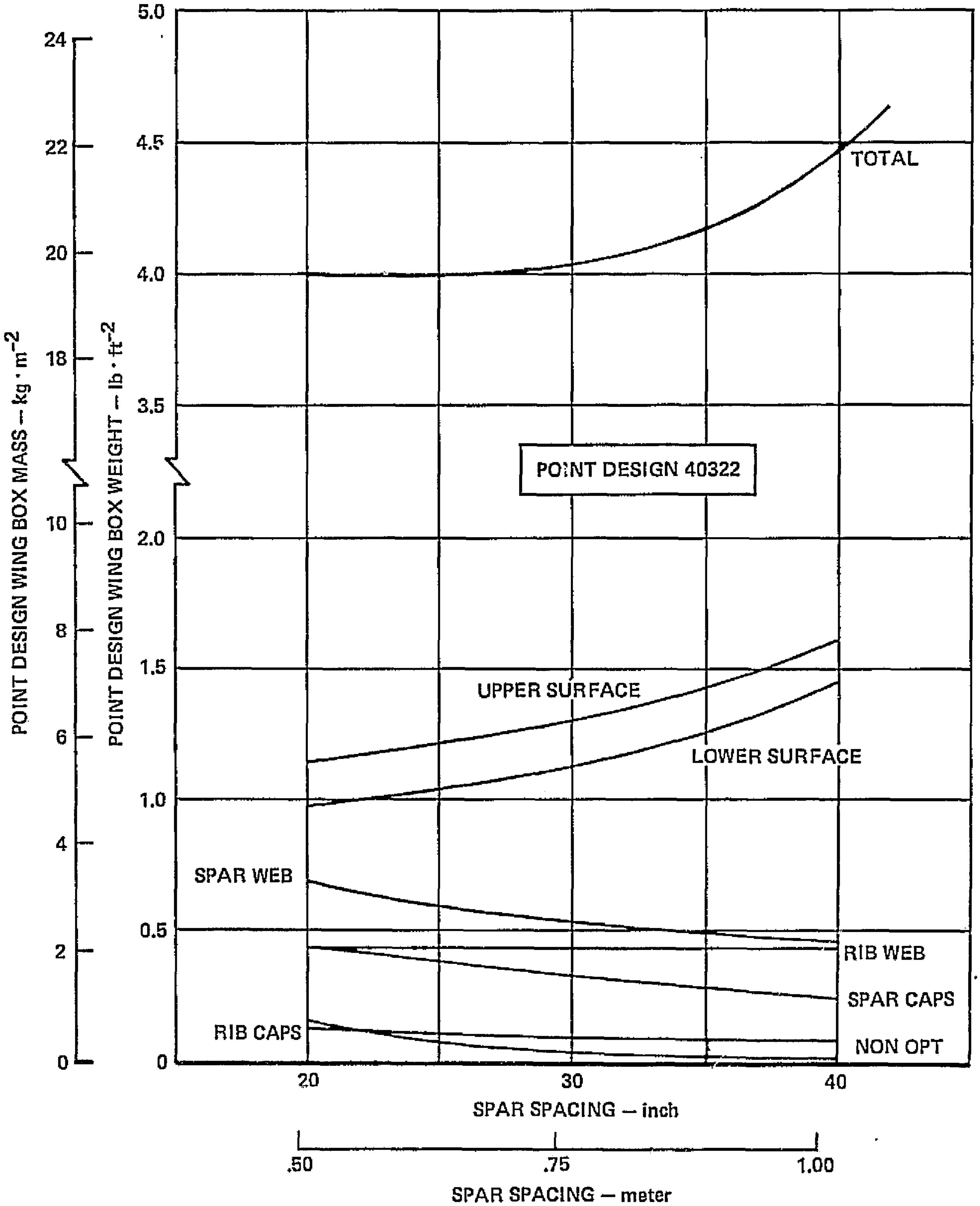


Figure 12-48. Optimum Spar Spacing for the Monocoque Mechanically Fastene: - Densified Core Arrangement, Point Design Region 40322

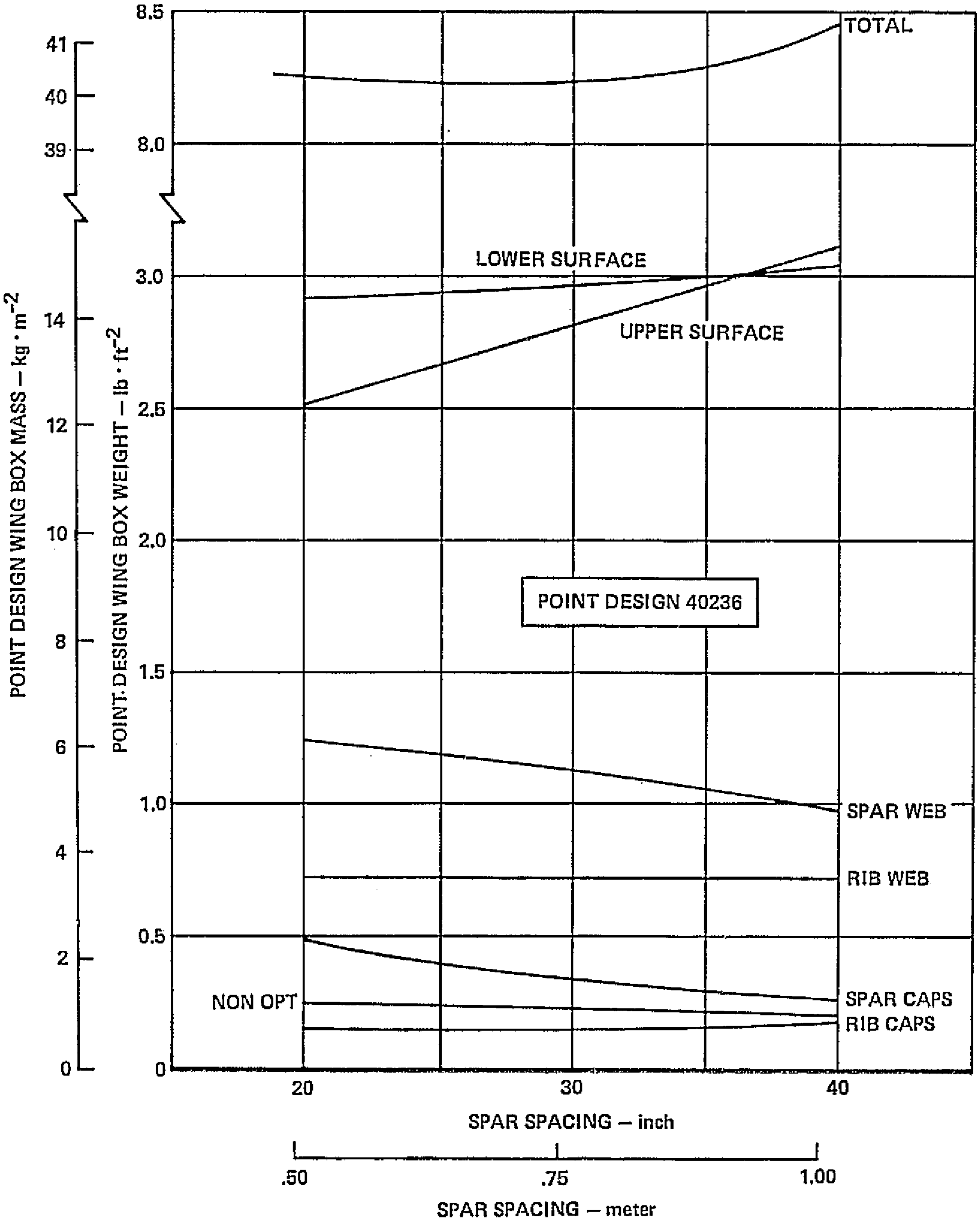


Figure 12-49. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Densified Core Arrangement, Point Design Region 40236

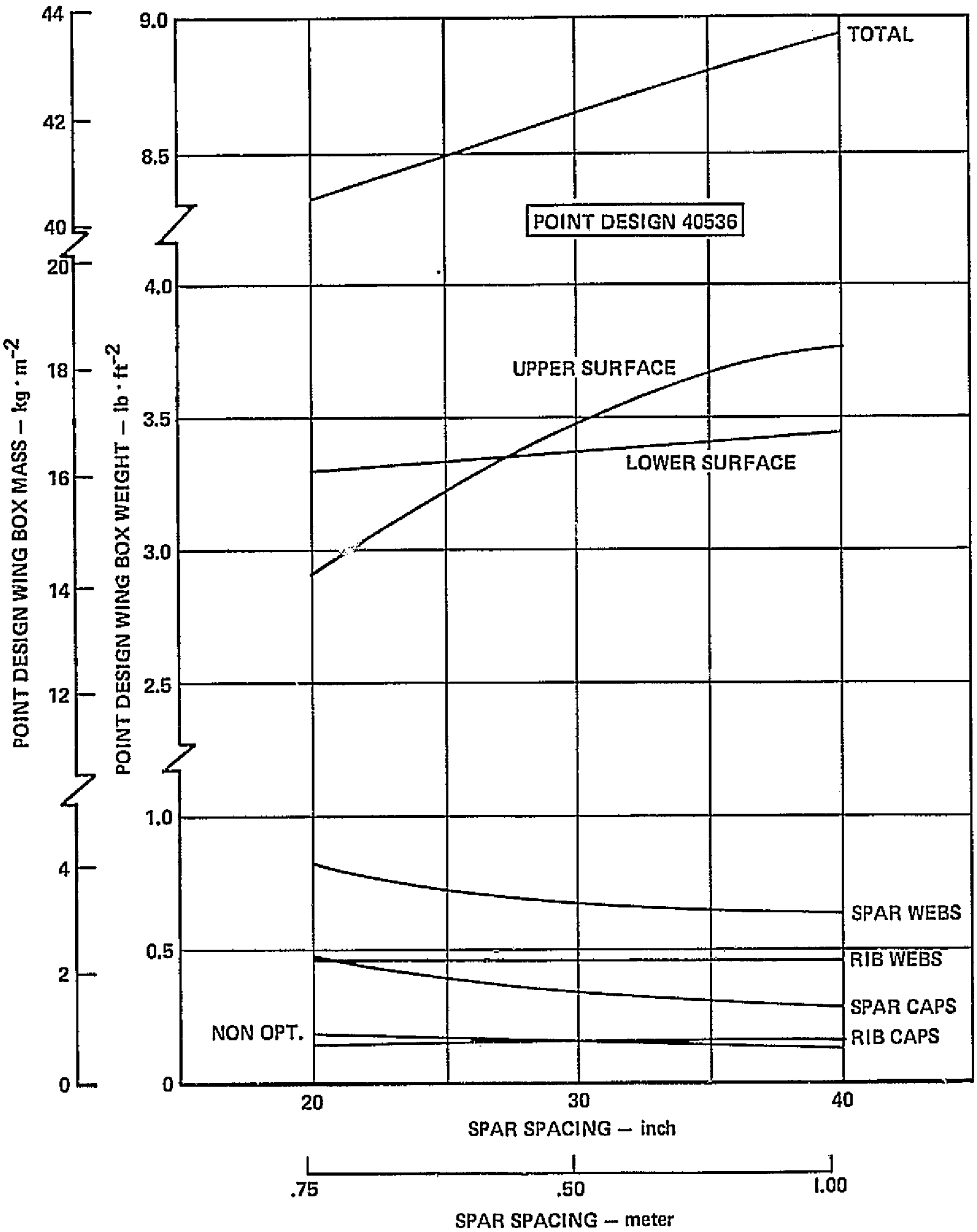


Figure 12-50. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Densified Core Arrangement, Point Design Region 40536

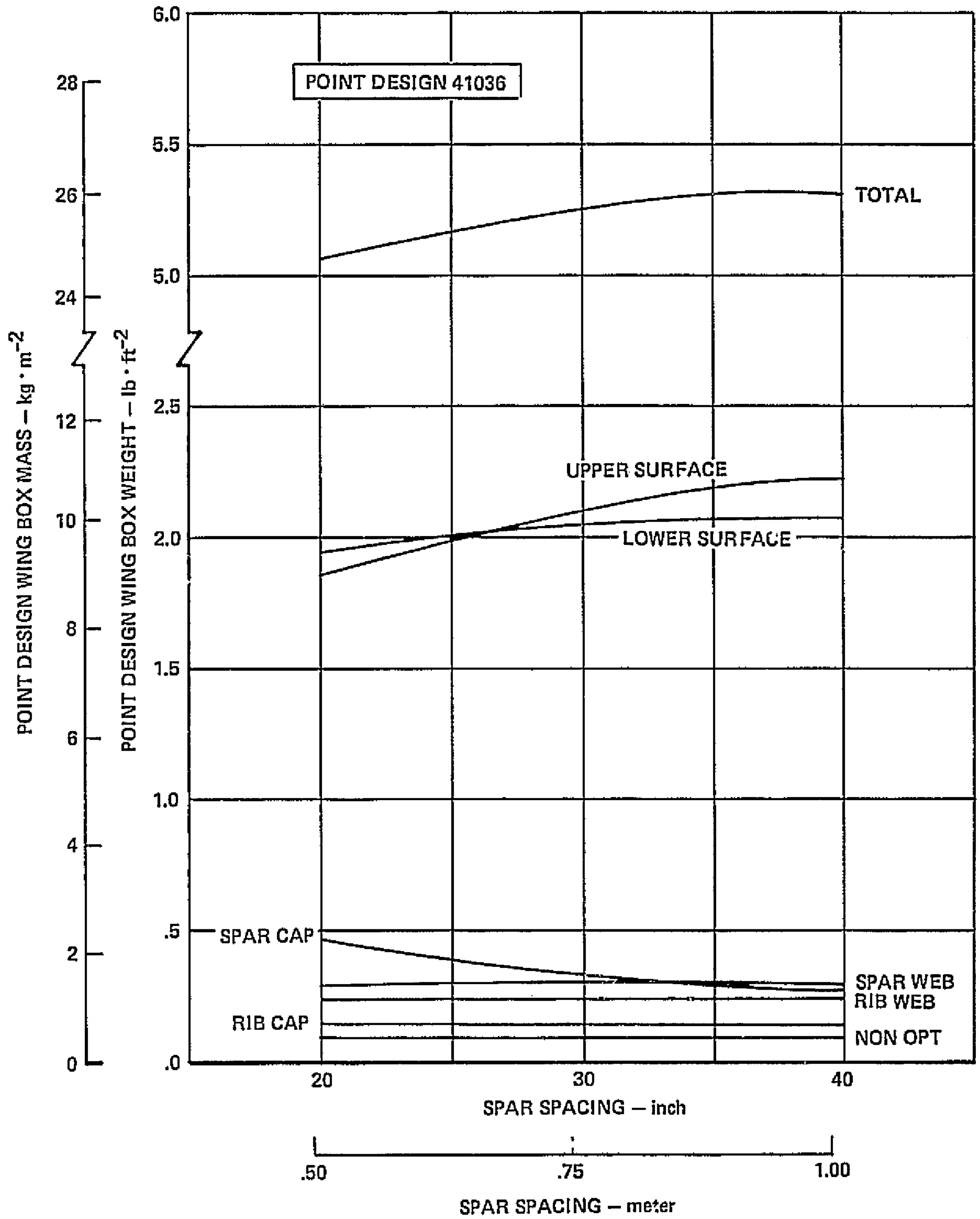


Figure 12-51. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Densified Core Arrangement, Point Design Region 41036

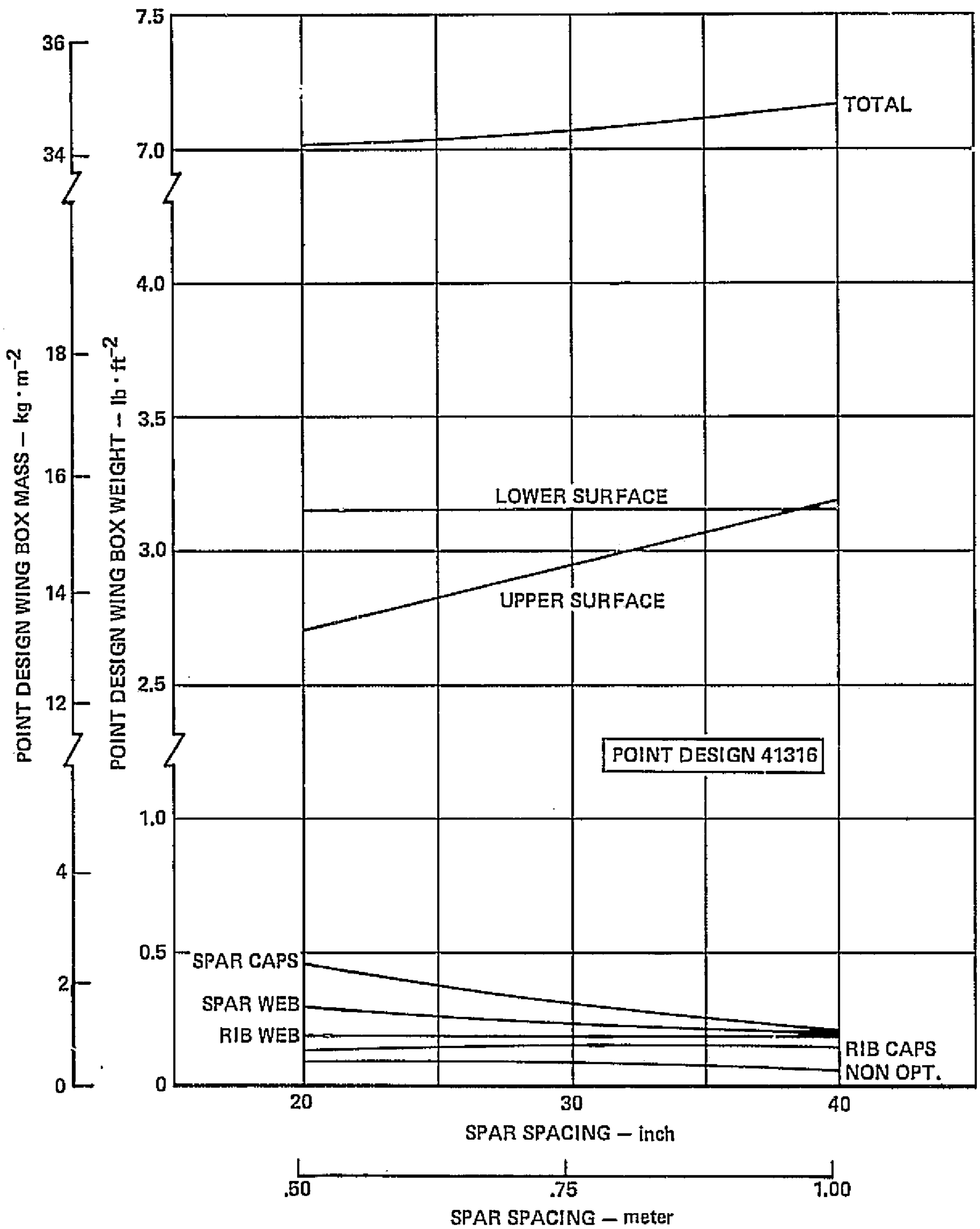


Figure 12-52. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Densified Core Arrangement, Point Design Region 41316

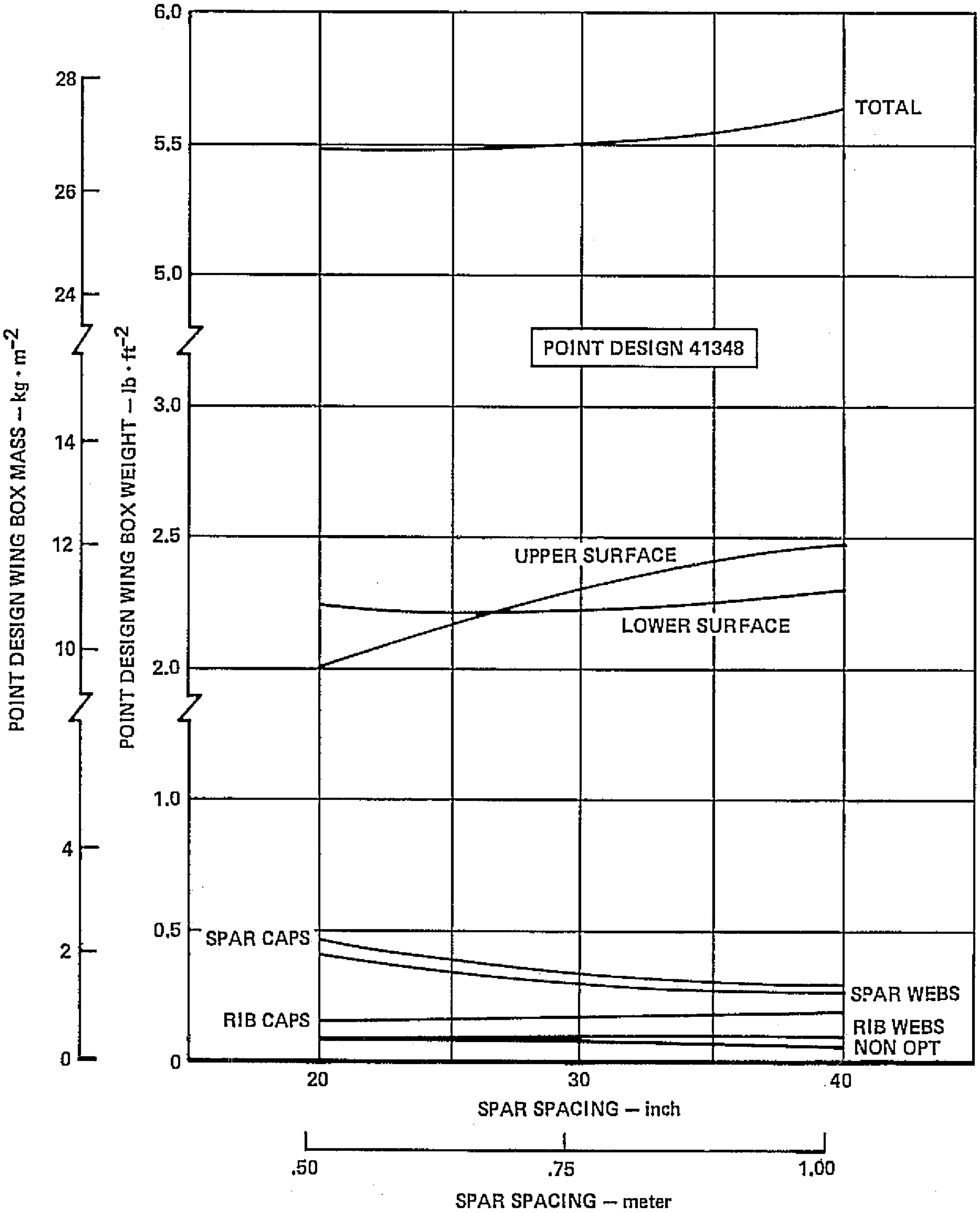


Figure 12-53. Optimum Spar Spacing for the Monocoque Mechanically Fastened - Densified Core Arrangement, Point Design Region 41348

TABLE 12-42. WEIGHT COMPARISON OF THE TASK I MONOCOQUE WING ARRANGEMENT

POINT DESIGN REGIONS	SPAR SPACING (IN.)	UNIT WING BOX WEIGHTS			
		MINIMUM — WEIGHT ARRANGEMENT (LB./SQ. FT.) (1)	NORMALIZED VALUES (2)		
			MECH. FAST. — TUBULAR	WELDED — TUBULAR	MECH. FAST. — DENSIFIED CORE
40322	20.0	4.00	1.13	1.09	1.00
40236	20.0	8.26	N.A. (3)	1.03	1.00
40536	20.0	8.33	1.04	1.03	1.00
41036	20.0	5.05	N.A. (3)	1.04	1.00
41316	20.0	7.02	N.A. (3)	1.05	1.00
41348	20.0	5.49	1.05	1.06	1.00

1. MINIMUM WEIGHT ARRANGEMENT ~ MECHANICALLY FASTENED — DENSIFIED CORE
2. ALL VALUES NORMALIZED TO THE MINIMUM-WEIGHT ARRANGEMENT
3. WEIGHT DATA NOT AVAILABLE (N.A.)

the greatest weight-saving potential were the metallic surface panels and submerged spar caps. Figure 12-54 shows these components and a typical wing box segment for a chordwise wing arrangement.

The composite reinforced designs were analyzed for the same point design environment as defined for the metallic chordwise wing arrangement. Table 12-2 contains the critical surface panel load-temperature environment. The composite reinforced concepts were subjected to the same combinations of load as the metallic concepts and sized for equal or greater strength. For the design loads the ultimate strength of the composite and titanium elements were not exceeded. For the tension condition, the ultimate tensile stress in the titanium substrate did not exceed the fatigue allowable. Under the application of the design loads the combined structure experience neither general instability nor local instability failures. The effect of thermal curing stresses and thermal gradients on the strength of the reinforced element was also evaluated.

The composite reinforced concepts were designed to have equal or greater stiffness than the representative metallic concepts used in the chordwise structural model. This criteria applies to both the shear stiffness (Gt) and axial stiffness (Et) of the section in the principal stiffness direction.

The reinforced concepts were sized for the same panel proportion (rib/spar spacing) studied for the chordwise metallic panels and the resulting least-weight panel geometry and associated rib/spar spacing were defined.

Composite Wing Surface Panels

A systematic approach was used to evaluate the application of composite to the chordwise stiffened wing panels. The initial task involved screening the composite material system and selecting the most promising material for a more in-depth study. Those materials considered were: Graphite/polyimide (Gr/PI), Boron/polyimide (B/PI) and Boron/aluminum (B/Al). Upon selecting the material system, a weight-strength analysis was conducted with variable spar spacing to define the panel proportions which exhibit the largest potential weight saving over the metallic design. Then using these panel proportions, the detail panel geometry was determined at each of the six point design regions.

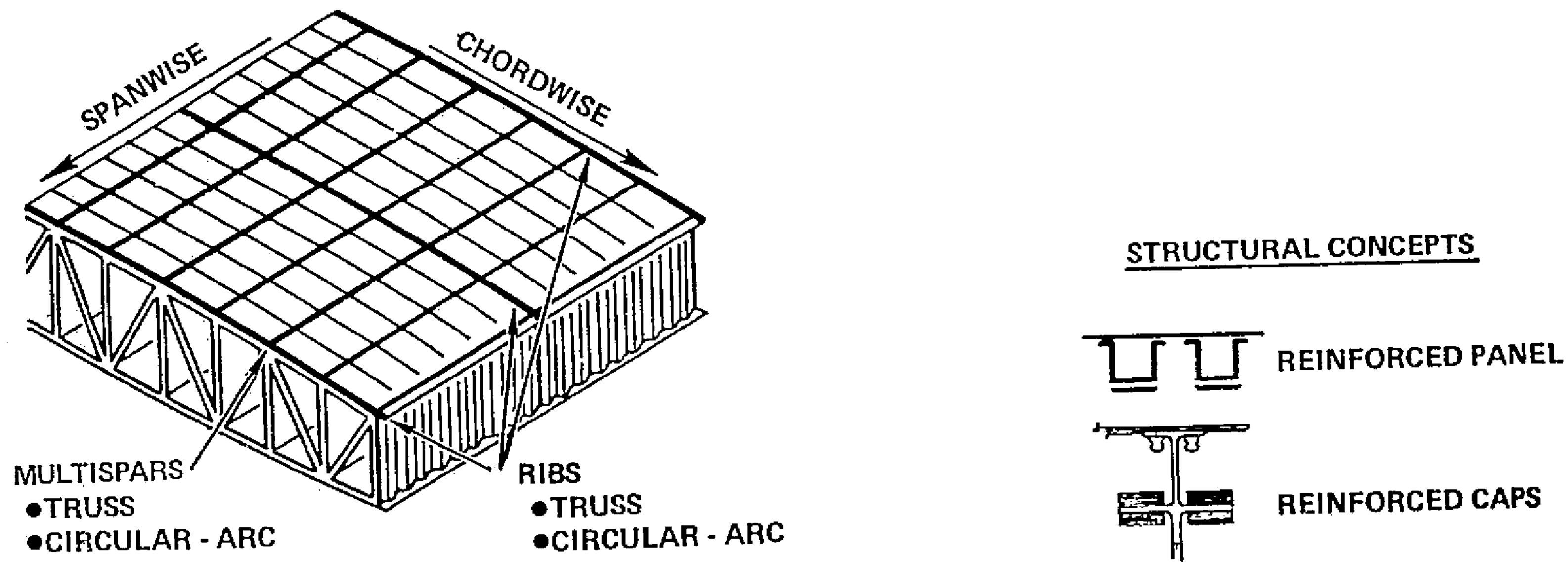


Figure 12-54. Composite Reinforced Wing Structural Arrangements

Material System Trade-off Study - The three candidate composite material systems evaluated were MODMOR II/Skybond 703 Graphite/polyimide (Gr/PI), Boron/Skybond 703 (B/PI), and 5.6 Boron/1100 Aluminum with titanium interleaves (B/Al). The static material properties for these materials are presented in Section 7 and are based on currently published (1970-1972 technology) data which have minimal statistical basis.

Comparative weight-strength studies were performed using the hat-stiffened concept to assess the relative weight of these candidate composite materials. This study included sizing both upper and lower surface panels for spar spacings of 20-, 30-, and 40-inches. A sample of these results are presented in Figure 12-55 for the hat-stiffened concept reinforced with each of the candidate composite materials. This data is for 20-inch spar spacing at point design region 40536 with the least-weight metallic chordwise design, convex-beaded concept, included for comparison purposes. These panel weight results indicated the following ranking for the candidate composite materials: (1) Graphite/polyimide, (2) Boron/polyimide, and (3) Boron/aluminum. The Graphite/polyimide reinforced design is the least-weight composite concept (3.21 lb/sq.ft.) indicating approximately 3-percent and 11-percent weight savings over the Boron/polyimide and Boron/aluminum designs, respectively. The metallic design, which weighs 2.94 lb/sq.ft., is approximately 8-percent lighter than the minimum-weight (Gr/PI) composite reinforced design.

Based on the results of the material trade-off study the Graphite/polyimide material system was selected as the most promising composite material for application to the surface panels.

Detail Panel Analysis - The Graphite/polyimide panel design was subjected to further analysis to ascertain the panel dimensions (rib/spar spacing) affording the greatest weight savings potential over the minimum-weight metallic design. Table 12-43 summarizes these results for point design regions 40536 and 41348 and includes the panel cross-section dimensions, unit weights, and critical design conditions for both upper and lower surface panels. The surface panel unit weight from this table are graphically displayed in Figure 12-56 with the corresponding data for the lightest weight metallic concept (convex-beaded). With regard to this figure, the weight of the composite reinforced panels are almost invariant with respect to spar spacing e.g., only a six-percent weight increase is indicated when the spacing is

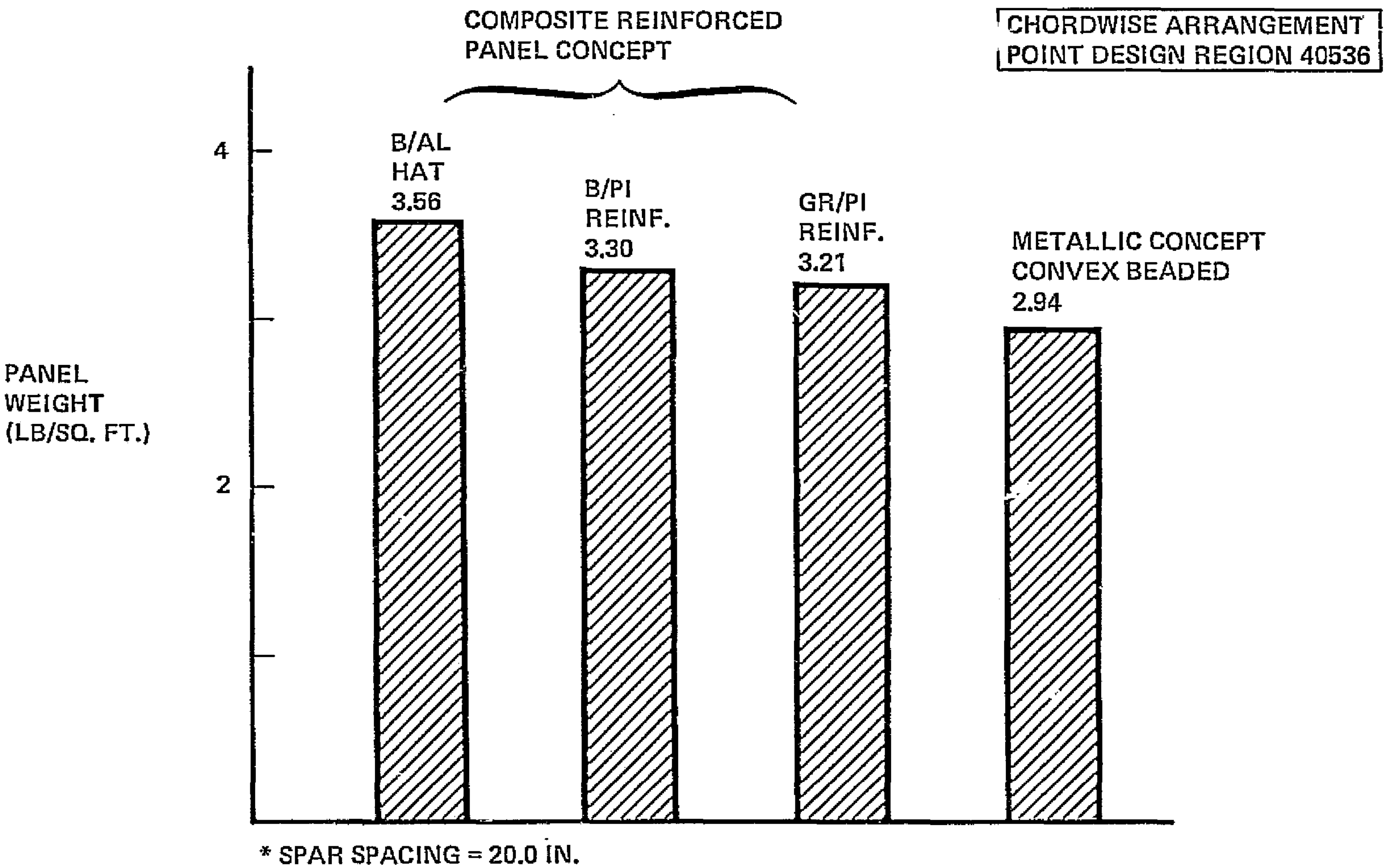
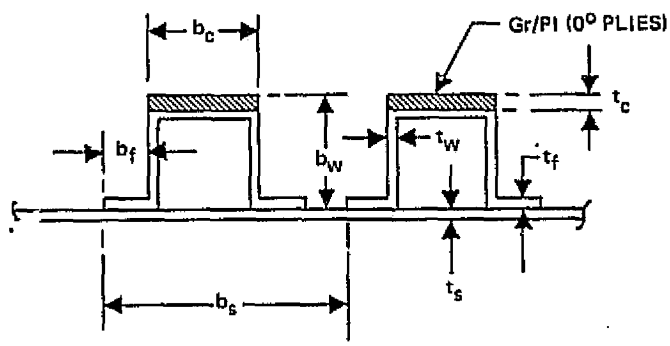


Figure 12-55. Weight Comparison of Candidate Composite Materials

TABLE 12-43. PANEL GEOMETRY AND WEIGHT FOR THE GR/PI COMPOSITE REINFORCED PANEL CONCEPT

POINT DESIGN REGION	40536						41348					
SURFACE	UPPER			LOWER			UPPER			LOWER		
SPAR SPACING (IN.)	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:												
t_s (IN.)	0.052	0.052	0.052	0.042	0.042	0.042	0.054	0.054	0.054	0.045	0.045	0.045
b_s (IN.)	3.200	3.300	3.100	3.100	2.900	2.500	3.300	3.300	3.100	3.200	3.100	3.300
b_f (IN.)	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400
b_w (IN.)	0.640	0.880	0.850	0.600	0.600	1.080	0.650	0.850	0.600	0.600	0.600	1.040
b_c (IN.)	1.500	1.550	1.450	1.450	1.350	1.150	1.550	1.550	1.450	1.500	1.450	1.550
t_f (IN.)	0.018	0.018	0.015	0.015	0.015	0.017	0.019	0.018	0.015	0.015	0.015	0.018
t_w (IN.)	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015
t_c (IN.)	0.055	0.065	0.070	0.015	0.030	0.040	0.055	0.065	0.070	0.025	0.030	0.040
MASS DATA:												
W (LB./FT.^{2})	1.806	1.886	1.958	1.402	1.469	1.536	1.831	1.903	1.978	1.491	1.515	1.539
CRITICAL CONDITION	31	31	31	31	31	31	31	31	31	31	31	31



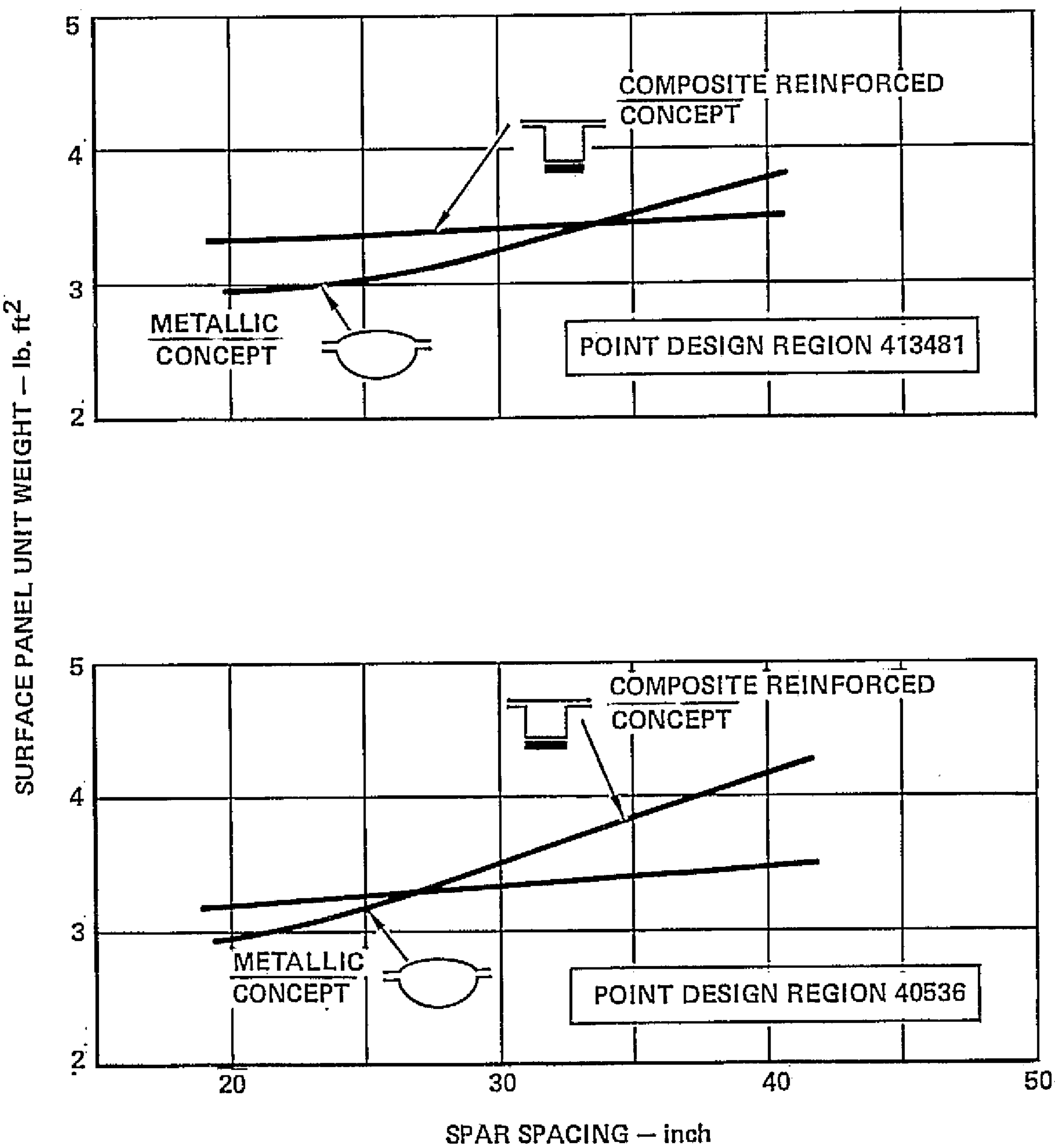


Figure 12-56. Optimum Spar Spacing for the Gr/PI Composite Reinforced Panel

increased from 20-inches to 40-inches at point design region 41348. Relative to the weight of the metallic panel designs, the composite reinforced panels are heavier for the 20-inch spar spacing designs, show no definitive weight trend for the 30-inch design, and indicate a decisive weight advantage for the 40-inch design. A weight savings of approximately 8-percent and 16-percent are noted for the 40-inch composite reinforced designs for regions 41348 and 40536, respectively. Since the largest weight savings are indicated for the larger spar spacings, the panel cross-section dimensions and unit weights for the 40-inch spar spacing are shown in Table 12-44 for all six wing point design regions.

Composite Substructure

Similar to the philosophy adopted for the composite surface panel, the application of composites to the substructure was restricted to reinforcing metallic designs. The major weight components were reviewed and the component exhibiting the greatest potential weight saving was selected for investigation; namely, the submerged spar caps of the chordwise wing arrangement e.g., the weight of the metallic spar caps for 20-inch spacing at region 40536 are approximately 60-percent of the total box weight.

For the Task I composite substructure analysis, only metallic spar caps with Boron/polyimide reinforcement were considered. Methods used to conduct the analysis and the results of the analysis are presented in the following section.

Methods - For the analysis of the composite reinforced metallic spar caps, allowable tension and compressive strength curves were defined. The basic material properties of the 6Al-4V Titanium alloy and the Boron/polyimide reinforcement are presented in Section 7, Table 7-3. The laminate (combined titanium and B/PI) tensile and compression stress-strain curves are presented in Figure 12-57 and 12-58 for various proportions (by cross-sectional area) of unidirectional Boron/polyimide. A curing thermal differential temperature of 300°F was assumed.

Figure 12-59 presents the allowable laminate tensile stress developed from the tensile stress-strain curves at the fatigue cutoff strength of the titanium alloy, 90,000 psi. For the allowable laminate compressive stress, the fiber failure point

TABLE 12-44. PANEL GEOMETRY AND WEIGHT FOR THE GR/PI COMPOSITE REINFORCED PANELS

POINT DESIGN REGION	40322		40236		40536		41036		41316		41348	
SURFACE	UP	LOW	UP	LOW	UP	LOW	UP	LOW	UP	LOW	UP	LOW
RIB SPACING (IN.)	40	40	40	40	40	40	40	40	40	40	40	40
DIMENSIONS:												
t_s (IN.)	0.026	0.031	0.036	0.048	0.052	0.042	0.053	0.046	0.078	0.096	0.054	0.045
b_s (IN.)	1.500	1.900	2.300	2.900	3.100	2.500	3.400	3.600	3.600	4.800	3.100	3.300
b_f (IN.)	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400	0.400
b_w (IN.)	0.670	0.940	0.610	0.930	0.850	1.080	0.960	0.690	0.610	0.850	0.600	1.040
b_c (IN.)	0.650	0.850	1.030	1.350	1.450	1.150	1.600	1.700	1.700	2.300	1.450	1.550
t_f (IN.)	0.015	0.015	0.015	0.015	0.015	0.017	0.018	0.015	0.015	0.015	0.015	0.018
t_w (IN.)	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015	0.015
t_c (IN.)	0.035	0.045	0.050	0.070	0.070	0.040	0.065	0.035	0.045	0.045	0.070	0.040
MASS DATA:												
W (LB./FT.^{2})	1.341	1.497	1.464	1.829	1.958	1.536	1.900	1.549	2.305	2.120	1.978	1.539
CRITICAL CONDITION	20	31	31	31	31	31	31	31	31	31	31	31

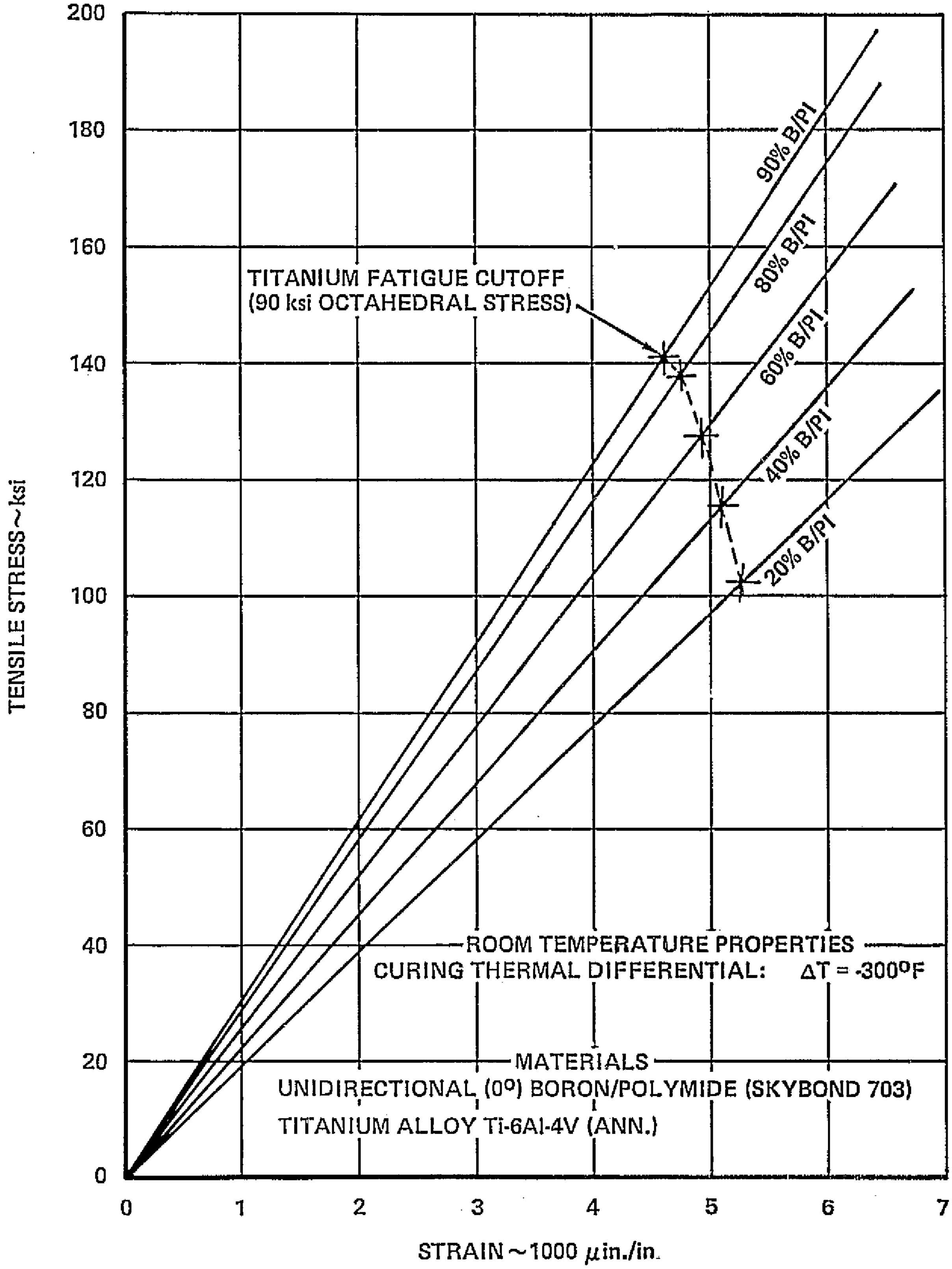
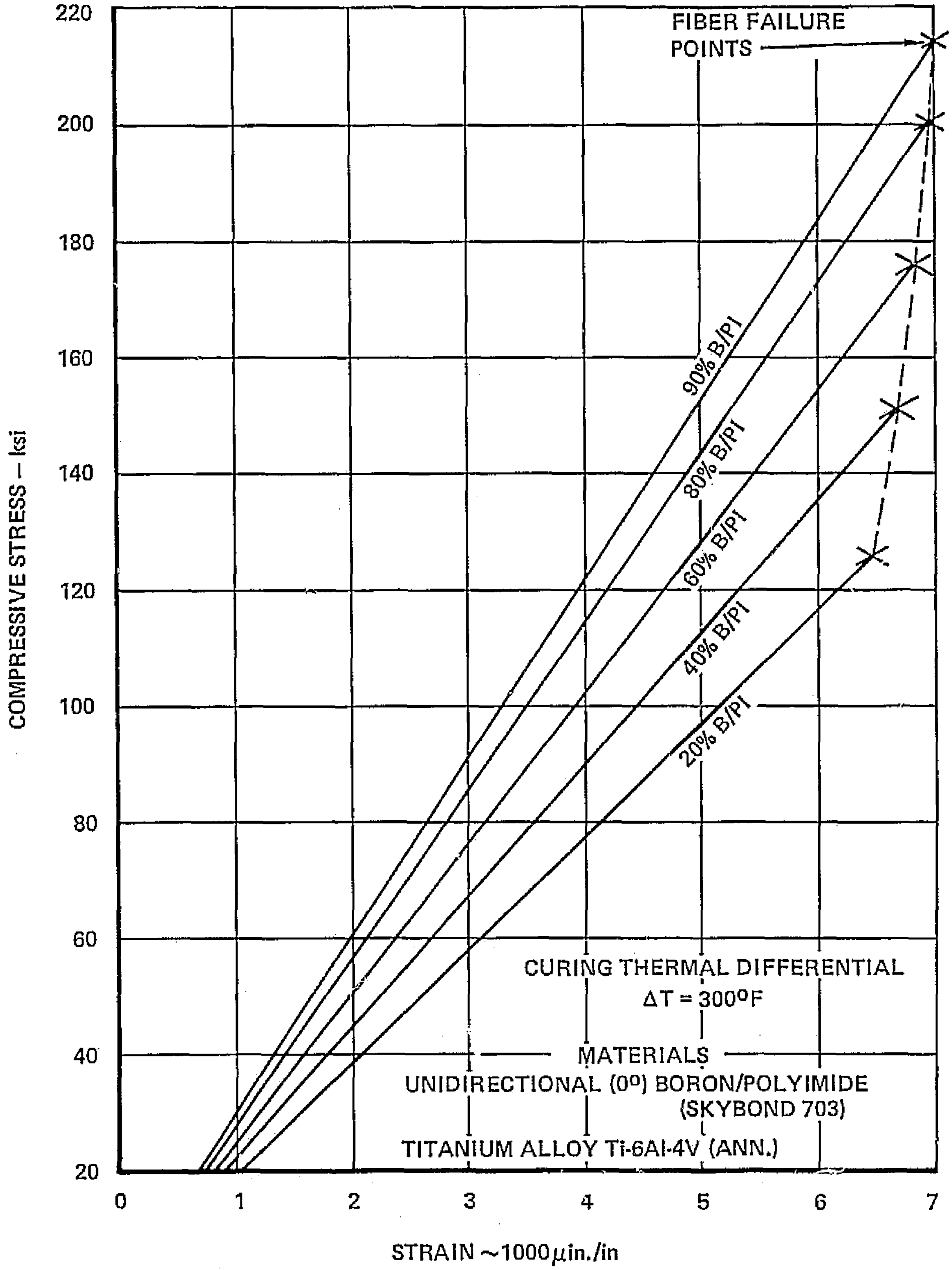


Figure 12-57. Tensile Strength of B/PI Reinforced Titanium



12-58. Compressive Strength of B/PI Reinforced Titanium

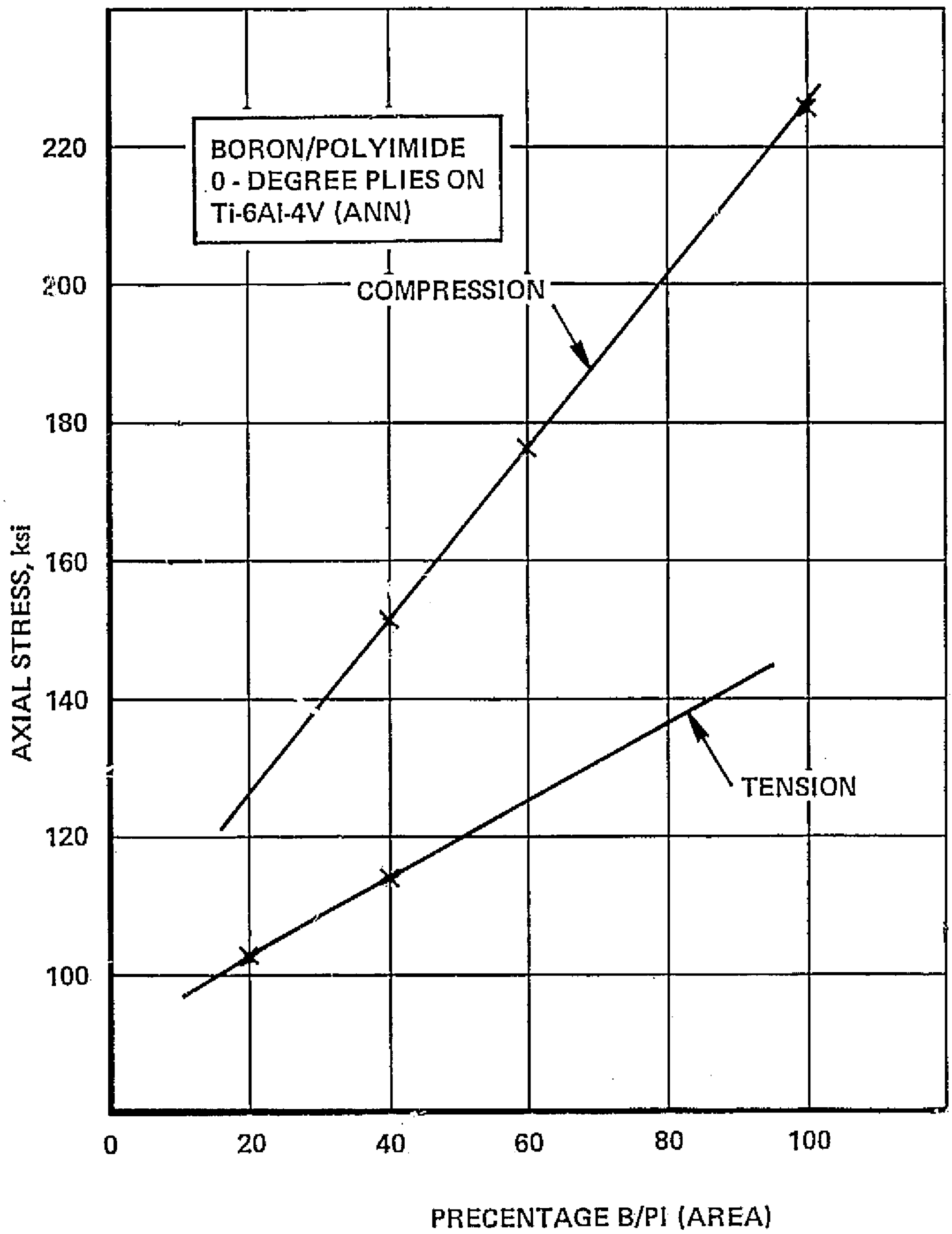


Figure 12-59. Critical Axial Stress for B/PI Reinforced Titanium

defines the maximum compressive stress for each proportion of Boron/polyimide. These stress levels are the terminus point of the stress-strain curves shown in Figure 12-58. These compressive allowables are superposed on the same figure used to present the tension allowables, Figure 12-59.

<u>Composite Spar Cap Analysis</u> - A sample of stress analysis performed on the B/PI reinforced spar caps at regions 40322, 40536, and 41348 are shown in Table 12-45. The same loads were used as for the metallic spar caps of the chordwise wing arrangement. Using these design loads (Table 12-11), the area of the metal and Boron/polyimide components were determined by using the allowable curves on Figure 12-59. Using a minimum design area for the titanium substrate, the percentage of B/PI (by cross-sectional area) was varied until the applied stress ($f^{C,T} = P_{ULT}/A_T$) approached the allowable stress ($F^{C,T}$). The resulting margins of safety are included on Table 12-45.

From the gross area proportions determined from the stress analysis, care was exercised in the distribution of this area into realistic dimensions to preclude any local or general instability failures. Tables 12-46 and 12-47 contain the spar cap dimensions for the six wing point design regions. In addition to presenting the area and dimensions, the equivalent surface panel unit weights are shown for each design, and the equations used for these calculations are presented in the footnotes of these tables.

The results of the composite reinforced spar cap analysis are summarized in Table 12-48 for each of the wing point design regions. Included on this table are the corresponding weights of the metallic design caps and the percentage weight saving afforded by the application of composite reinforcement to the spar caps. In general, large weight saving are indicated for the composite reinforced designs in the highly loaded wing regions i.e., aft box and wing tip. A minimum weight savings of 52-percent is noted for the upper surface cap at point design region 41036 and a maximum weight saving of 69-percent for the upper surface cap at region 41346. For the lightly loaded forward wing box region (40322), no appreciable weight saving over the all metal titanium design was noted for the 20-inch design caps; whereas, the composite reinforced designs for the 40-inch spar spacing offers a 28-percent and 44-percent weight saving over the corresponding upper and lower surface caps of the titanium design.

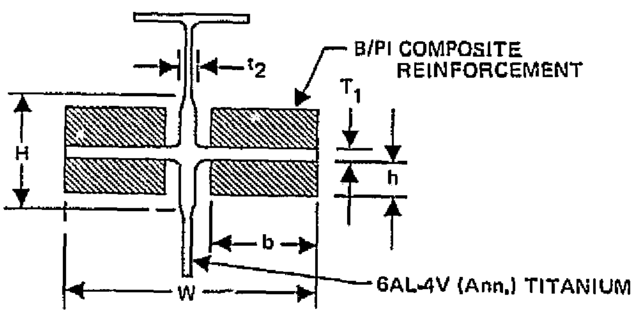
TABLE 12-45. COMPOSITE (B/PI) REINFORCED SPAR CAP ANALYSIS

POINT DESIGN REGION	SURFACE	SPAR SPACING (IN.)	APPLIED LOAD(1) COND. NO.	APPLIED LOAD(1) P_{ULT} (KIPS)	AREA(2) A_M	AREA(2) A_C	AREA(2) A_T	$f^{C,T}$(3) (KSI)	% COMPOSITE	$F^{C,T}$(4) (KSI)	M.S.(5)
40322	UPPER	20	9	-27.4	0.21	—	0.21	-131.1	—	-131.0	0.00
		40	9	-56.0	0.24	0.16	0.40	-140.0	40	-151.0	0.08
	LOWER	20	9	27.4	0.30	—	0.30	90.0	—	90.0	0.00
		40	9	56.0	0.24	0.25	0.49	114.3	51	120.0	0.05
40536	UPPER	20	31	-307.4	0.45	1.20	1.65	-186.3	73	-193.0	0.04
		40	31	-656.6	0.45	2.78	3.23	-203.3	86	-208.0	0.02
	LOWER	20	31	307.4	0.45	1.96	2.41	+127.6	81	136.0	0.06
		40	31	656.6	0.45	4.71	5.16	+127.2	91	141.0	0.11
41348	UPPER	20	31	-272.2	0.45	1.04	1.49	-182.7	70	-188.0	0.03
		40	31	-632.0	0.45	2.60	3.05	-207.2	85	-207.0	0.00
	LOWER	20	31	272.2	0.45	1.68	2.13	127.8	79	136.0	0.06
		40	31	632.0	0.45	4.41	4.86	130.0	91	141.0	0.08

1. ULTIMATE LOADS PER TABLE 12-11.
2. CAP AREAS:
   A_M = METALLIC AREA
   A_C = COMPOSITE AREA
   A_T = TOTAL ARC
3. APPLIED STRESS (ULT.) $f^{C,T} = P_{ULT.}/A_T$
4. ALLOWABLE STRESS PER FIGURE 12-59
5. MARGIN OF SAFETY (M.S.) = $(F^{C,T}/f^{C,T}) - 1$

TABLE 12-46. GEOMETRY AND WEIGHT OF THE COMPOSITE (B/PI) REINFORCED SPAR CAPS

POINT DESIGN REGION	SURFACE	SPACING (IN.)	SPAR CAP DIMENSIONS h (IN.)	b (IN.)	H (IN.)	W (IN.)	t_1 (IN.)	t_2 (IN.)	AREAS A_M (IN.^{2})	A_C (IN.^{2})	UNIT WEIGHT w (LB./SQ.FT.)
40322	UPPER	20	—	—	—	—	—	—	—	—	—
		40	0.08	0.50	1.00	1.50	0.09	0.11	0.24	0.16	0.18
	LOWER	20	—	—	—	—	—	—	—	—	—
		40	0.12	0.50	1.00	1.50	0.09	0.11	0.24	0.25	0.20
40536	UPPER	20	0.30	1.00	1.20	2.50	0.12	0.13	0.45	1.20	1.14
		40	0.56	1.25	1.20	3.00	0.10	0.13	0.45	2.78	0.98
	LOWER	20	0.49	1.00	1.20	2.50	0.12	0.13	0.45	1.96	1.53
		40	0.67	1.75	1.20	4.00	0.08	0.13	0.45	4.71	1.48
41348	UPPER	20	0.26	1.00	1.20	2.50	0.12	0.13	0.45	1.04	1.06
		40	0.52	1.25	1.20	3.00	0.10	0.13	0.45	2.60	0.94
	LOWER	20	0.42	1.00	1.20	2.50	0.12	0.13	0.45	1.68	1.38
		40	0.63	1.75	1.20	4.00	0.08	0.13	0.45	4.41	1.41



w = EQUIVALENT UNIT SURFACE PANEL WEIGHT

$= (\rho_M A_M + \rho_C A_C) \times 144/a$; LB./SQ.FT.)

WHERE

ρ_M = METAL DENSITY (ti6A1-4V)
= 0.16 LB./IN.^{3}

A_M = METAL AREA
$= (H - t_1)t_2 + Wt_1$;IN.^{2}

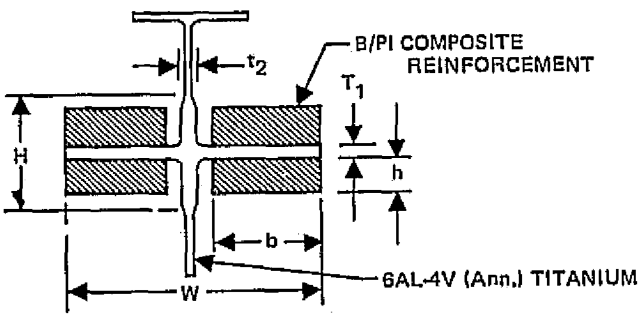
ρ_C = COMPOSITE DENSITY (B/PI)
= 0.072 LB./IN.^{3}

AC = COMPOSITE AREA = b x h x 4; IN.^{2}

a = SPAR SPACING

TABLE 12-47. GEOMETRY AND WEIGHT OF THE COMPOSITE (B/PI) REINFORCED SPAR CAPS

POINT DESIGN REGION	SURFACE	SPAR SPACING (IN.)	SPAR CAP DIMENSIONS h (IN.)	b (IN.)	H (IN.)	W (IN.)	t_1 (IN.)	t_2 (IN.)	AREAS A_M (IN.^{2})	A_C (IN.^{2})	UNIT WEIGHT w (LB/SQ. FT.)
40236	UPPER	20	0.35	1.00	1.20	2.50	0.12	0.13	0.45	1.40	1.24
		40	0.64	1.25	1.20	3.00	0.10	0.13	0.45	3.20	1.07
	LOWER	20	0.59	1.00	1.20	2.50	0.12	0.13	0.45	2.36	1.74
		40	0.79	1.75	1.20	4.00	0.08	0.13	0.45	5.53	1.70
41036	UPPER	20	0.18	1.00	1.20	2.50	0.12	0.13	0.45	0.72	0.90
		40	0.38	1.25	1.20	3.00	0.10	0.13	0.45	1.90	0.75
	LOWER	20	0.29	1.00	1.20	2.50	0.12	0.13	0.45	1.16	1.12
		40	0.45	1.75	1.20	4.00	0.08	0.13	0.45	3.15	1.07
41316	UPPER	20	0.44	1.00	1.20	2.50	0.12	0.13	0.45	1.76	1.44
		40	0.87	1.25	1.20	3.00	0.10	0.13	0.45	4.35	1.38
	LOWER	20	0.74	1.00	1.20	2.50	0.12	0.13	0.45	2.96	2.05
		40	1.07	1.75	1.20	4.00	0.08	0.13	0.45	7.49	2.20



w = EQUIVALENT UNIT SURFACE PANEL WEIGHT

$= (\rho_M A_M + \rho_C A_C) \times 144/a$; LB./SQ.FT.

WHERE:

ρ_M = METAL DENSITY (Ti6A1-4V) = 0.16 LB./IN.^{3}

A_M = METAL AREA = $(H - t_1)t_2 + Wt_1$; IN.^{2}

ρ_C = COMPOSITE DENSITY (B/PI) = 0.072 LB./IN.^{3}

A_C = COMPOSITE AREA = $b \times h \times 4$; IN.^{2}

a = SPAR SPACING

TABLE 12-48. WEIGHT COMPARISON OF THE METALLIC AND COMPOSITE REINFORCED SPAR CAPS

POINT DESIGN REGION	SURFACE	SPAR SPACING (IN.)	SPAR CAP DESIGN: COMPOSITE REINFORCED		SPAR CAP DESIGN: ALL METAL	SPAR CAP WEIGHT SAVING (PERCENT)
			UNIT WEIGHT (LB./SQ. FT.)	PERCENT COMPOSITE	UNIT WEIGHT (LB./SQ. FT.)	
40322	UPPER	20	–	–	0.24	–
		40	0.18	23	0.25	28
	LOWER	20	–	–	0.35	–
		40	0.20	32	0.36	44
40236	UPPER	20	1.24	58	3.16	61
		40	1.07	78	3.31	68
	LOWER	20	1.74	70	4.75	63
		40	1.70	84	4.97	66
40536	UPPER	20	1.14	54	2.71	58
		40	0.98	74	2.89	66
	LOWER	20	1.53	41	3.95	52
		40	1.48	82	4.19	65
41036	UPPER	20	0.90	41	1.87	52
		40	0.75	66	2.08	64
	LOWER	20	1.12	54	2.71	59
		40	1.07	76	3.02	64
41316	UPPER	20	1.44	63	3.92	63
		40	1.38	82	4.50	69
	LOWER	20	2.05	75	5.73	64
		40	2.20	88	6.55	66
41348	UPPER	20	1.06	51	2.41	56
		40	0.94	72	2.78	66
	LOWER	20	1.38	63	3.48	60
		40	1.41	81	4.06	65

Composite Wing Box

Wing box weights for the chordwise arrangement were investigated for the application of composite to both panels (Gr/PI) and spar caps (B/PI), and for spar caps only. For both applications the remaining structural weights corresponded to the metal designs as previously discussed for the chordwise wing arrangement.

A comparison of the box weights of the two composite arrangements with the least-weight metallic arrangement is presented in Figure 12-60 for point design region 40536. Both composite designs, composite reinforced panels and spar caps and the application of composites to the spar caps only, afforded weight saving of approximately 35 percent over the all metallic designs for comparable spar spacings.

With respect to the composite reinforced arrangements, the arrangement which incorporated the composites reinforced spar caps was least-weight for the 20-inch spar spacing; whereas, the arrangement which incorporated both reinforced panels and spars afforded the least-weight design for the larger 40-inch spar spacing. Detail weight statements for these two composite reinforced arrangements are shown in Tables 12-49 and 12-50. The detail weights for the metallic components were as present in the chordwise arrangement analysis; whereas, the detail component weights for the composite reinforced structure were as presented in Tables 12-44 and 12-48.

FUSELAGE STRUCTURAL ARRANGEMENT - TASK I

The major fuselage structural components (panels and frame) were subjected to point design analysis commensurate with the stages of design incorporated in the Task I analytical design studies, these stages were:

- Initial Screening - A preliminary parametric frame spacing study to ascertain the spacing associated with minimum weight design; then using this spacing, a structural analysis was performed to screen the fuselage panel candidates to determine the most promising concept or combination of concepts for further evaluation.
- Detailed Concept Analysis - A detail analysis of the surviving concept(s) from the Initial Screening analysis.

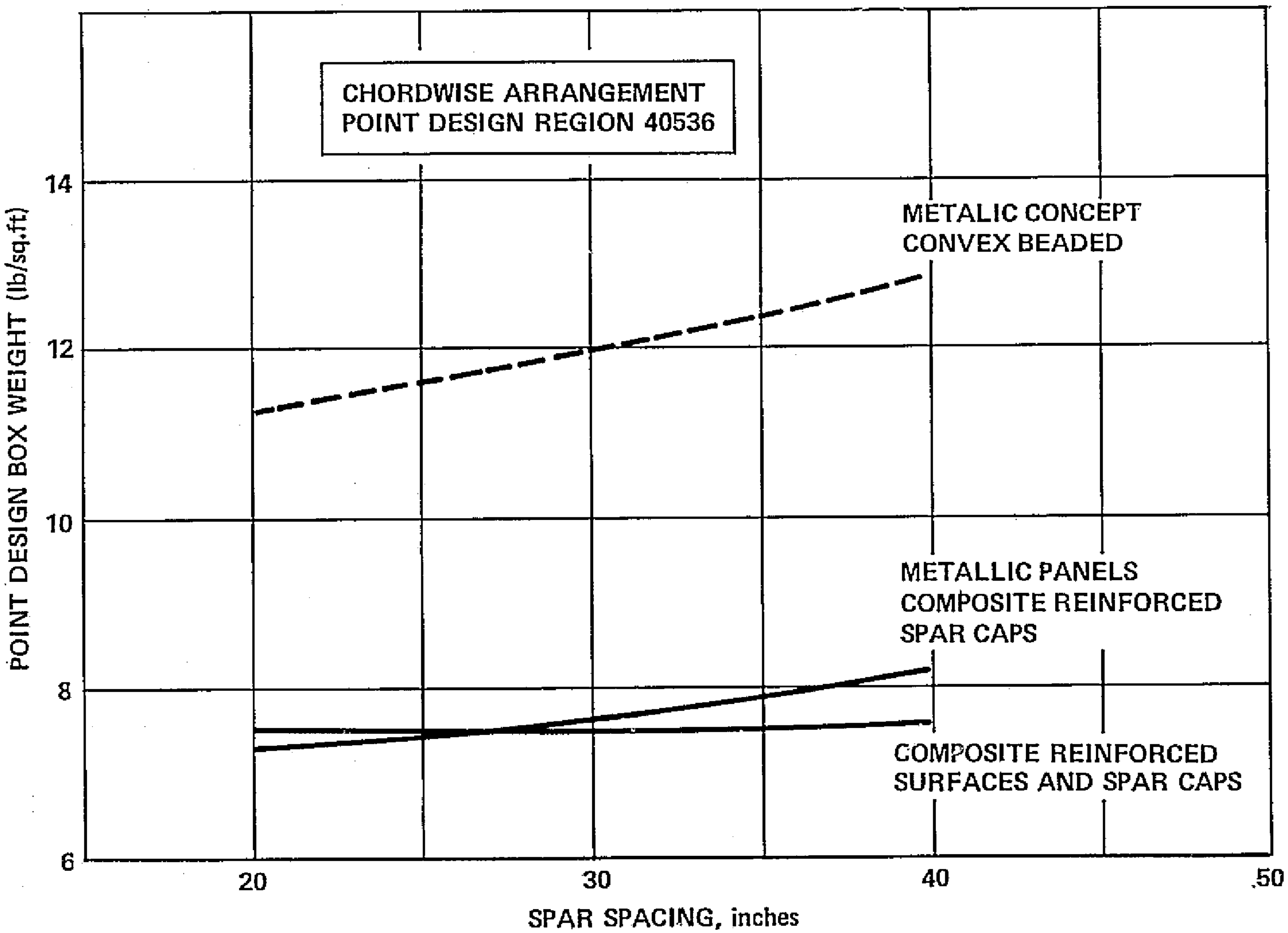


Figure 12-60. Optimum Spar Spacing for the Composite Reinforced Wing Arrangements

TABLE 12-49. DETAIL WING WEIGHTS FOR THE CHORDWISE ARRANGEMENT WITH COMPOSITE REINFORCED SPAR CAPS

POINT DESIGN REGION	40322	40236	40536	41036	41316	41348
SPAR SPAC (IN.)	20	20	20	20	20	20
PANELS						
UPPER	0.825	1.032	1.609	1.452	2.571	1.632
LOWER	0.942	1.279	1.335	1.320	2.007	1.366
Σ	(1.767)	(2.311)	(2.944)	(2.772)	(4.578)	(2.998)
RIB WEBS						
BULKHEAD	0.298	0.279	0.238	0.111	0.270	0.106
TRUSS	0.074	0.237	0.228	0.060	—	—
Σ	(0.372)	(0.516)	(0.466)	(0.171)	(0.270)	(0.106)
SPAR WEBS						
BULKHEAD	0.336	0.361	0.270	0.109	0.439	0.291
TRUSS	0.301	0.544	0.490	0.359	—	—
Σ	(0.637)	(0.905)	(0.760)	(0.468)	(0.439)	(0.291)
RIB CAPS						
UPPER	0.058	0.070	0.116	0.093	0.160	0.103
LOWER	0.065	0.083	0.086	0.087	0.126	0.074
Σ	(0.123)	(0.153)	(0.202)	(0.180)	(0.286)	(0.177)
SPAR CAPS						
UPPER	0.241	1.240	1.140	0.900	1.440	1.060
LOWER	0.350	1.740	1.530	1.120	2.050	1.380
Σ	(0.591)	(2.980)	(2.670)	(2.020)	(3.490)	(2.440)
NON-OPTIMUM						
MECH. FAST.	0.180	0.200	0.200	0.200	0.200	0.200
WEB INTERS.	0.120	0.120	0.120	0.120	0.120	0.120
Σ	(0.300)	(0.320)	(0.320)	(0.320)	(0.320)	(0.320)
Σ POINT DESIGN MASS $\frac{LB}{FT^2}$	3.790	7.180	7.360	5.930	9.380	6.330

TABLE 12-50. DETAIL WING WEIGHTS FOR THE CHORDWISE ARRANGEMENT WITH COMPOSITE REINFORCED SURFACE PANELS AND SPAR CAP

POINT DESIGN REGION	40322	40236	40536	41036	41316	41348
SPAR SPAC (IN.)	40	40	40	40	40	40
PANELS						
UPPER	1.341	1.464	1.958	1.900	2.305	1.978
LOWER	1.497	1.829	1.536	1.549	2.120	1.539
Σ	(2.838)	(3.293)	(3.494)	(3.449)	(4.425)	(3.517)
RIB WEBS						
BULKHEAD	0.298	0.279	0.238	0.111	0.270	0.106
TRUSS	0.074	0.237	0.228	0.060	–	–
Σ	(0.372)	(0.516)	(0.466)	(0.171)	(0.270)	(0.106)
SPAR WEBS						
BULKHEAD	0.336	0.451	0.375	0.151	0.288	0.192
TRUSS	0.153	0.323	0.325	0.201	–	–
Σ	(0.489)	(0.774)	(0.700)	(0.352)	(0.288)	(0.192)
RIB CAPS						
UPPER	0.097	0.099	0.130	0.120	0.167	0.129
LOWER	0.073	0.112	0.116	0.091	0.141	0.088
Σ	(0.170)	(0.211)	(0.246)	(0.211)	(0.308)	(0.217)
SPAR CAPS						
UPPER	0.180	1.070	0.980	0.750	1.380	0.940
LOWER	0.200	1.700	1.480	1.070	2.200	1.410
Σ	(0.380)	(2.770)	(2.460)	(1.820)	(3.580)	(2.350)
NON-OPTIMUM						
MECH. FAST.	0.160	0.180	0.180	0.180	0.180	0.180
WEB INTERS.	0.100	0.100	0.100	0.100	0.100	0.100
Σ	(0.260)	(0.280)	(0.280)	(0.280)	(0.280)	(0.280)
Σ POINT DESIGN MASS $\frac{LB}{FT^2}$	4.510	7.840	7.650	6.280	9.150	6.660

For these design studies, analyses were conducted on a point design basis at four discrete fuselage locations. These locations are shown in Figure 12-4 overlayed on a planform view of the arrow-wing configuration and included fuselage stations 750, 2000, 2500, and 3000.

The structural arrangements investigated in the Task I studies included conventional skin-stringer and frame designs. For the panels, zee-and hat-stiffened concepts were investigated, with both open and closed designs considered for the hat-stiffened concept. A floating frame with skin shear-ties was the only candidate considered for frame design. These candidate concepts were previously shown in Figure 12-2.

Point design analyses were conducted for both the Initial Screening and Detailed Concept Analyses on the aforementioned structural concepts at the selected fuselage regions. The specific load/temperature environment, methods, and analysis are included in the discussion for each stage of design.

As specified in the point design environment (Section 11), since the Task I structural model contained a coarse fuselage model, all Task I internal loads were based on existing loads from references 4 and 5. These external loads are presented in Figures 12-61 and 12-62 where the maximum point design values for FS 2000, FS 2500, and FS 3000 are:

FS	BENDING MOMENT (IN-LBS)	SHEAR (LB)
2000	150×10^6	300×10^3
2500	200×10^6	350×10^3
3000	150×10^6	300×10^3

Internal loads were defined for each stage of the Task I analyses using the above applied loads and theoretical bending (MC/I) and shear (VQ/I) distributions.

Frame Spacing Study

A study was conducted to define the frame spacing associated with minimum-weight fuselage design. For this study, a simplified weight-strength analysis was conducted on each of the three panel candidates to establish their weight trend as a function of frame spacing. The single frame design was included in this analysis and was invariant for all panel concepts.

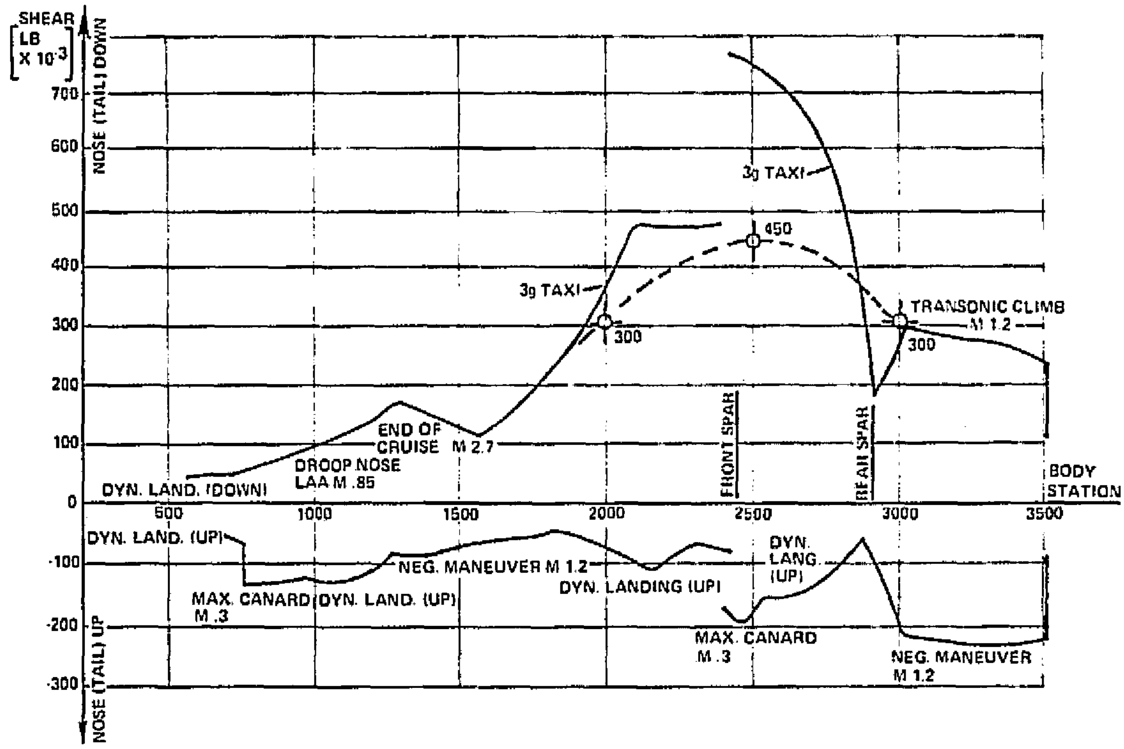


Figure 12-61. Fuselage Shear Diagram - Task I

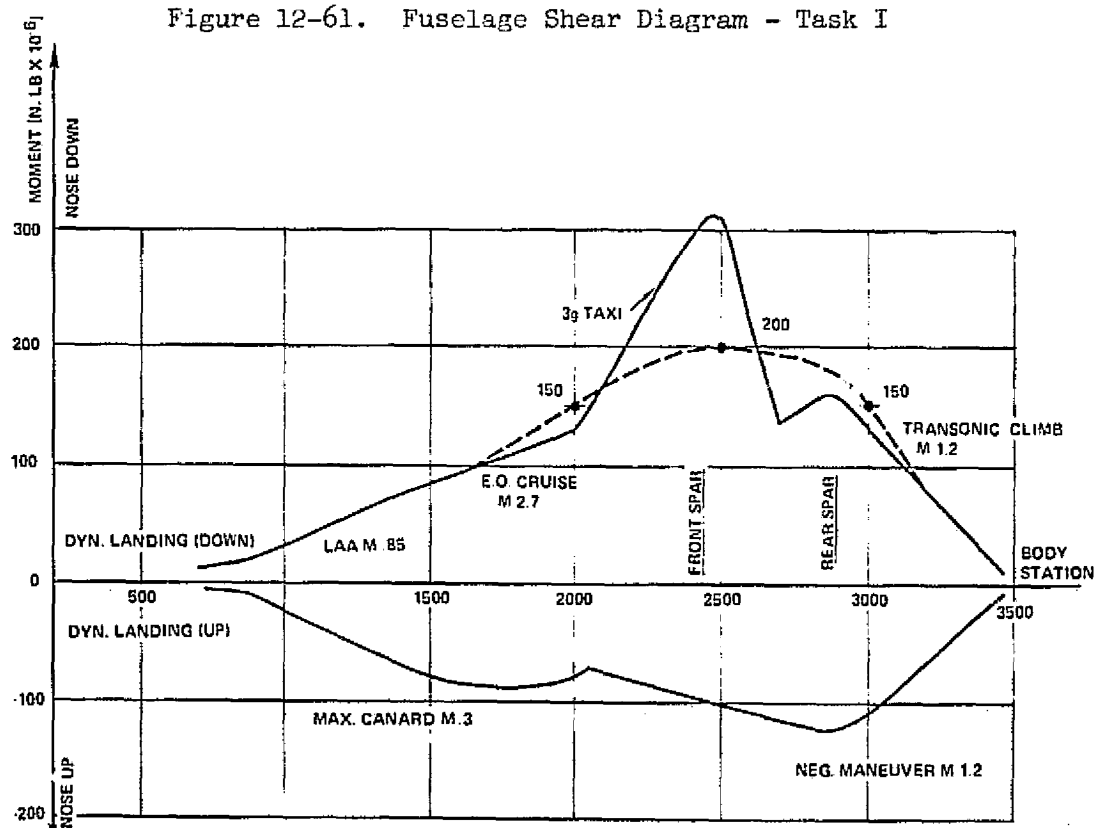


Figure 12-62. Fuselage Bending Moment Diagram - Task I

Panel Sizing - The panel inplane loads were determined using the applied bending moments and shears as previously specified in Figures 12-61 and 12-62. Table 12-51 contains a summary of the internal forces resulting using the theoretical MC/I and VQ/I distributions. A maximum axial load of 17,600 lb/in. occurs at the extreme fibers of FS 2500 where the corresponding maximum shear flow on the side panel is 2100 lb/in. Discrete panels at each cross-section were analyzed for failure under combined compression (tension) and shear loadings. The maximum normal stresses (f_n) were calculated using the principal stress equation:

$$f_n = \frac{f_x}{2} \pm \sqrt{\left(\frac{f_x}{2}\right)^2 + f_{xy}^{\ 2}}$$

where the axial stress (f_x) and shear stress (f_{xy}) represents the stress intensity normal and parallel to the surface, respectively.

The panel margins of safety were determined by comparing the stresses calculated by the above equation with the appropriate allowable stress.

The allowable compressive stresses and corresponding panel geometry were determined by the theory defined by Emero and Spunt in Reference 2, i.e., wide column allowables.

For fuselage bending material, the ultimate design gross area stress in tension was limited to 90,000 psi, see fuselage fatigue analysis, Section 13. For this tension condition, the principal stresses were calculated using the optimum panel cross-section geometry for compression design panels. An example of the results of this analysis are shown in Table 12-52. This table summarizes the results of the hat-stiffened panel analysis at FS 2500, and includes the equivalent panel thicknesses for frame spacings of 10-, 20-, and 30-inches. For the maximum tension loads, upper location, a constant panel thickness of 0.196-inch is noted, whereas, for the maximum compression loaded fibers the thicknesses range from 0.155- to 0.187-inch for the three frame spacings investigated.

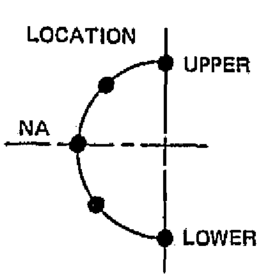
A comparison of the panel thicknesses for the candidate concepts at each of the three point design region are shown in Table 12-53. The hat-stiffened concepts (open and closed) have approximately the same weight which is lighter (approximately five-percent) than the zee-stiffened designs all point design regions.

TABLE 12-51. FUSELAGE PANEL LOAD INTENSITIES, FRAME SPACING STUDY - TASK I

LOCATION	FUSELAGE PANEL LOAD INTENSITIES (ULT.), LB/IN			
	DIRECTION	FS 2000	FS 2500	FS 3000
UPPER PANEL	N_x	13200	17600	13200
	N_{xy}	170	255	170
SIDE PANEL	N_x	0	0	0
	N_{xy}	1400	2100	1400
LOWER PANEL	N_x	-13200	-17600	-13200
	N_{xy}	170	255	170

TABLE 12-52. RESULTS OF FUSELAGE PANEL ANALYSIS AT FS 2500

POINT DESIGN REGION	CIRCUM LOCATION	APPLIED LOADS (ULT.)		EQUIVALENT PANEL THICK. ($IN.^2/IN.$)		
		N_x (LB./IN.)	N_{xy} (LB./IN.)	L = 10 (IN.)	L = 20 (IN.)	L = 30 (IN.)
FS 2500	UPPER	17,600	255	.196	.196	.196
	UP – 45°	10,000	1,680	.129	.129	.129
	N.A.	0	2,100	.051	.072	.089
	LOW – 45°	-10,000	1,680	.092	.121	.150
	LOWER	-17,600	255	.155	.158	.187
	AVERAGE $\bar{t}$			.112	.125	.140



$$\text{AVERAGE } \bar{t} = \sum_{i=1}^{5} C_i \bar{t} \Big/ \sum_{i=1}^{5} C_i$$

L = FRAME SPACING

N_x = AXIAL LOAD

N_{xy} = SHEAR LOAD

Frame Sizing - The sizing of the frames for this parametric study were based on the theory derived by Shanley in Reference 6, which is premised on providing sufficient frame stiffness to preventing general instability of the shell in bending. Shanley's expression for the required frame stiffness is:

$$(EI) = C_f M \frac{D^2}{L}$$

This expression relates the frame stiffness (EI) to the applied shell bending moment (M), shell diameter (D) and the frame spacing (L). In addition, the recommended value of $1/16 \times 10^3$ was used for the frame stiffness coefficient (C_f). For this parametric study a constant thickness channel section frame 3.0 inches deep with constant width flanges of 1.0-inch was evaluated. For this cross section, a simplified expression was determined which relates the frame area to the frame moment of inertia, A = 0.74I. Using the above expression for frame stiffness with the assumed cross section relationship the required area, as a function of frame spacing, was defined for each point design region. Table 12-54 presents the results of this analysis conducted at FS 2500. For this point design region, the equivalent panel thickness of the frame ranged from 0.104-inch for a spacing of 10-inches to a 0.012-inch thickness for the 30-inch frame spacing.

A comparison of frame equivalent panel thickness at each point design region is shown in Table 12-55. In general, the frame equivalent thicknesses for the 30-inch frame spacing are approximately 10-percent of the thickness values for the 10-inch spacing. The equivalent panel thickness for 20-inch frame spacing is .020-inch at FS 2000 and FS 3000, and .026-inch at FS 2500.

Results - The results of the panel and frame analyses were combined to indicate the fuselage weight trends at the three point design regions investigated. Figures 12-63 and 12-64 present this data for FS 2500, and FS 2000 and 3000, respectively. These figures present the component weights (panel and frame) and total weight, expressed as equivalent panel thickness, of the fuselage as a function of frame spacing. Figure 12-63 indicates a minimum weight design of approximately .15-inch is attainable for the hat-stiffened design at frame spacing between 20- to 25-inches. The corresponding minimum weight design, hat-stiffened panel concept, at FS 2000 and FS 3000 is approximately .12-inch for frame spacings between 20- and 30-inches.

TABLE 12-53. WEIGHT COMPARISON OF THE CANDIDATE FUSELAGE PANEL CONCEPTS

POINT DESIGN REGION	PANEL CONCEPT	EQUIVALENT THICKNESS, $\bar{t}$		
		L=10	L=20	L=30
FS 2000	HAT-STIFF	0.083	0.099	0.112
	ZEE-STIFF	0.087	0.103	0.117
FS 2500	HAT-STIFF	0.112	0.125	0.140
	ZEE-STIFF	0.116	0.130	0.145
FS 3000	HAT-STIFF	0.083	0.099	0.112
	ZEE-STIFF	0.087	0.103	0.117

TABLE 12-54. RESULTS OF FUSELAGE FRAME ANALYSIS AT FS 2500

POINT DESIGN REGION	FUSELAGE BENDING MOMENT M, (IN-LBS)	SHELL DIAMETER D, (IN.)	FRAME MODULUS E, (PSI)	FRAME STIFFNESS PARAMETER C_f	FRAME SPACING L, (IN.)	FRAME AREA A, (IN.)	EQUIV. PANEL THICKNESS $\bar{t}$, (M^2/IN.)
FS 2500	200×10^6	134.0	16×10^6	1/16000	10	1.04	0.104
	200×10^6	134.0	16×10^6	1/16000	15	0.69	0.046
	200×10^6	134.0	16×10^6	1/16000	20	0.52	0.026
	200×10^6	134.0	16×10^6	1/16000	25	0.42	0.017
	200×10^6	134.0	16×10^6	1/16000	30	0.35	0.012

EQUATIONS:

$$(EI) = C_f MD^2/L$$

FOR A/I = 0.74 (SEE ASSUMED CROSS SECTION)

$$A = 0.74\ (EI)$$

TABLE 12-55. FUSELAGE FRAME WEIGHTS, TASK I FRAME SPACING STUDY

POINT DESIGN REGION	BENDING MOMENT M, (IN-LB)	SHELL DIAMETER D, (IN.)	EQUIVALENT PANEL THICKNESS (IN^2/IN)		
			L=10	L=20	L=30
FS 2000	150×10^6	134.0	0.078	0.020	0.009
FS 2500	200×10^6	134.0	0.104	0.026	0.012
FS 3000	150×10^6	134.0	0.078	0.020	0.009

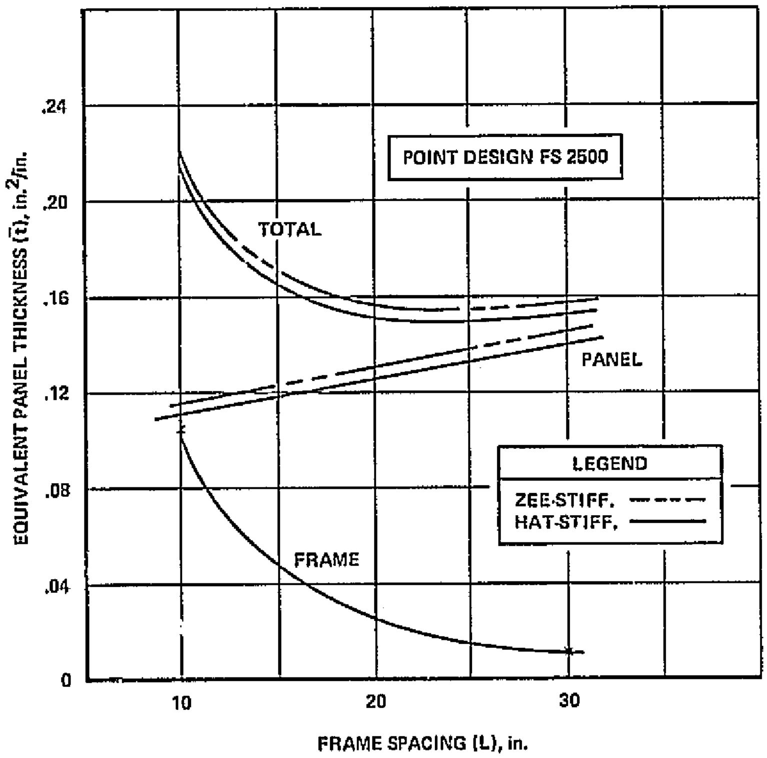


Figure 12-63. Optimum Frame Spacing for the Candidate Fuselage Arrangements, Point Design Region FS 2500

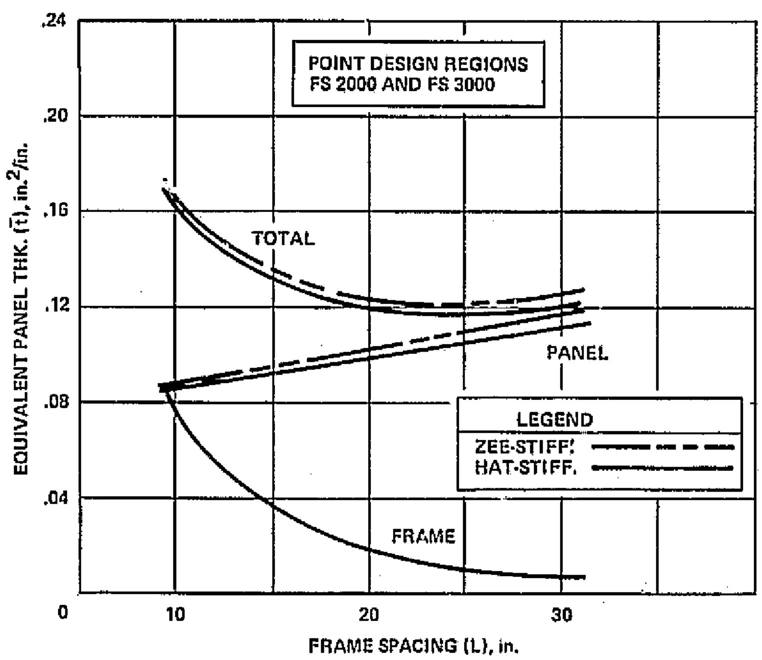


Figure 12-64. Optimum Frame Spacing for the Candidate Fuselage Arrangements, Point Design Regions FS 2000 and FS 3000

The results of the frame spacing study indicated frame spacings between 20- and 25-inches offer minimum-weight design. When these results are reviewed in conjunction with the results of the wing study, the lower bound value (20.0-inches) appears to be the most realistic spacing. Table 12-56 contains a weight comparison of the zee-and hat-stiffened designs for 20-inch frame spacing. The minimum-weight design hat-stiffened concept weighs 2.74 lb/sq.ft. at FS 2000 and FS 3000, and 3.48 lb/sq.ft. at FS 2500. The corresponding values for the zee-stiffened concept are approximately 3-percent higher.

Fuselage Initial Screening

To screen the fuselage panel concepts, a weight-strength analysis was conducted at the four point design region using the results of the prior frame spacing study, i.e., minimum-weight fuselage designs were indicated for 20-inch frame spacing.

Fuselage panel load intensities, axial load and shear flow, were calculated using the theoretical bending and shear distributions as previously discussed. For these calculations, the design loads (bending moment and shear) shown in Figures 12-61 and 12-62 were used in combination with the section properties defined in the frame spacing study. The point design environment included only the inplane load resulting from the fuselage bending and shear loads, internal pressure and temperatures were not considered for this screening investigation. Table 12-57 presents a summary of the fuselage panel load intensities at Fuselage Stations 2000, 2500, and 3000 for the maximum compression (lower panel), maximum shear (side panel), and the maximum tension (upper panel) panel locations.

For the stress analyses, the initial step was obtaining the gross area section properties and stresses, and the resulting load intensities. This data was adjusted until realistic stress levels were obtained. For example, when the tensile stress exceeded the fatigue allowable (90,000 psi), the equivalent panel thickness was increased until the stress level was equal to or lower than the allowable. These resulting load intensities were used for the detail stress analysis. For this analysis, the principal stress was calculated and compared to the applicable tension or compression allowable stress.

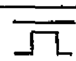
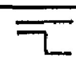
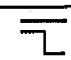
For the tension condition, the principal stress was compared to a gross area fatigue allowable stress of 90,000 psi. Similarly, the principal compressive stress was

TABLE 12-56. WEIGHT COMPARISON OF THE CANDIDATE FUSELAGE ARRANGEMENTS, TASK I FRAME SPACING STUDY

POINT DESIGN REGION	HAT-STIFF CONCEPTS			ZEE-STIFF CONCEPT		
	FRAME SPACING (IN.)	TOTAL $\bar{t}$ (IN.^{2}/IN.)	TOTAL W (LB/SQ. FT)	FRAME SPACING (IN.)	TOTAL $\bar{t}$ (IN.^{2}/IN.)	TOTAL W (LB/SQ. FT)
FS 2000	20.0	0.119	2.74	20.0	0.123	2.83
FS 2500	20.0	0.151	3.48	20.0	0.156	3.59
FS 3000	20.0	0.119	2.74	20.0	0.123	2.83

1. $\bar{t}$ = EQUIVALENT PANEL THICKNESS; $\bar{t}$ (FRAME) + $\bar{t}$ (PANEL)
2. W = EQUIVALENT PANEL WEIGHT; 23.04 X TOTAL $\bar{t}$

TABLE 12-57. FUSELAGE PANEL LOAD INTENSITIES, TASK I INITIAL SCREENING

LOCATION	FUSELAGE PANEL LOAD INTENSITIES (ULT.), LB/IN						
	DIRECTION	FS 2000, FS 3000			FS 2500		
							
UPPER PANEL	N_x	11600	11700	11600	15700	14600	15690
	N_{xy}	412	417	413	629	597	629
SIDE PANEL	N_x	377	406	300	422	545	416
	N_{xy}	1361	1357	1330	2025	2000	1998
LOWER PANEL	N_x	-11700	-11650	-12000	-16100	-16800	15900
	N_{xy}	415	412	426	645	670	633

compared to the most critical instability failure mode, i.e., either local or general instability. Figure 12-65 shows the allowable loads for a specific geometry hat-stiffened panel which has a constant stiffener geometry and thickness, and a variable skin thickness.

Table 12-58 contains a sample of the stress analysis conducted at the fuselage aft-body region, FS 3000. This table presents the panel cross section geometry, the applied and allowable stresses, and the margin of safety. The footnotes contain a sketch showing the circumferential location of the panels. A comparison of panel geometry is shown in Table 12-59 for each of the panel concepts. This geometry and weight comparison is made on the uppermost circumferential panels at each point design region.

The average panel weights for each fuselage region are shown in Table 12-60. The zee-stiffened concept is the lightest weight concept at FS 750 with an average weight of 1.31 lb/sq.ft. For the higher loaded regions - FS 2000, 2500, and 3000, the results of the analysis provided the following ranking of the panel concepts: closed hat-stiffened concept, open hat-stiffened concept and the zee-stiffened concept. This ranking was invariant at each of the regions with unit weights of 2.80 lb/sq.ft. and 3.18 lb/sq.ft. indicated for the least-weight closed hat-stiffened concept at FS 2000 and 3000, and FS 2500, respectively.

The fuselage component weights, frames and panels, and total weights are presented in Table 12-61 for each of candidate panel concepts. This data reflects the minimum-weight frame spacing of 20-inches and the frame weights (Table 12-55) ascertained in the previously described frame spacing study. Since the frame weights were invariant with each panel concept, the total weight (frame plus panel) reflects the same weight trend and hence have the same ranking as previously described when comparing the panel weights. The least weight fuselage concept, closed-hat stiffened concept, has a total unit weight of 3.26, 3.78, and 3.26 lb/sq.ft. at FS 2000, 2500, and 3000, respectively. The minimum total unit weight at FS 750 is 1.56 lb/sq.ft. for the zee-stiffened concept.

Fuselage Detailed Concept Analysis

The most promising structural concepts surviving the initial screening analysis were subjected to a more detailed analysis to refine the weight of the major

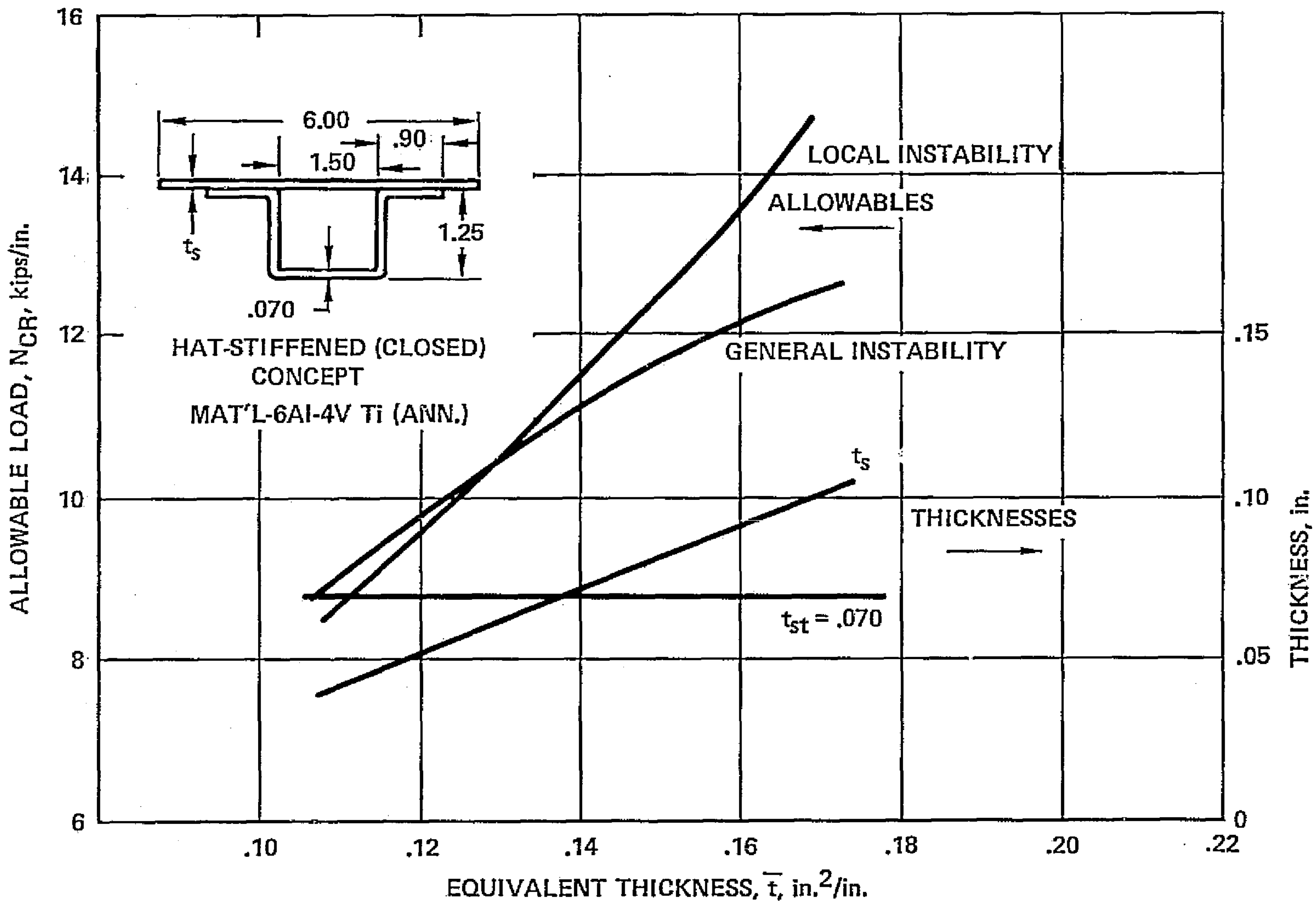
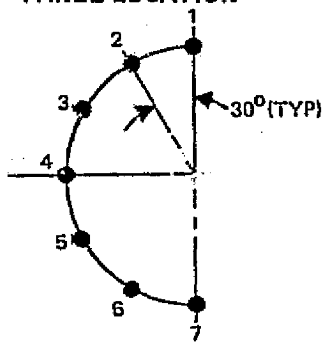


Figure 12-65. Panel Allowables for the Hot-Stiffened Panel Concept

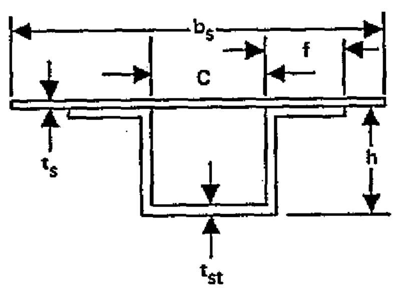
TABLE 12-58. RESULTS OF THE FUSELAGE PANEL ANALYSIS AT FS 3000, TASK I INITIAL SCREENING

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	FUSELAGE PANEL DIMENSIONS							APPLIED STRESS		ALLOWABLE STRESS F (KSI)	M.S.
			b_s (IN.)	t_s (IN.)	c (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	$\bar{t}$ (IN.^{2}/IN.)	f_x (KSI)	f_{xy} (KSI)		
FS 3000	CLOSED HAT	1	6.00	.080	1.50	.80	1.25	.063	.142	81.7	5.2	90.0	0.10
		2	6.00	.070	1.50	.75	1.25	.050	.120	71.9	7.6	90.0	0.24
		3	6.00	.063	1.50	.75	1.25	.040	.100	43.1	17.9	90.0	0.81
		4	6.00	.063	1.50	.75	1.25	.040	.100	3.8	21.6	90.0	LARGE
		5	6.00	.070	1.50	.75	1.25	.050	.120	-35.5	19.0	-55.2	0.26
		6	6.00	.080	1.50	.80	1.25	.063	.142	-64.3	16.2	-75.7	0.11
		7	6.00	.090	1.50	.90	1.25	.070	.159	-73.6	4.6	-76.0	0.03

PANEL LOCATION



PANEL CROSS SECTION



HAT-STIFFENED (CLOSED) CONCEPT

MARGIN OF SAFETY (M.S.)

$$MS = \frac{F}{f_n} - 1$$

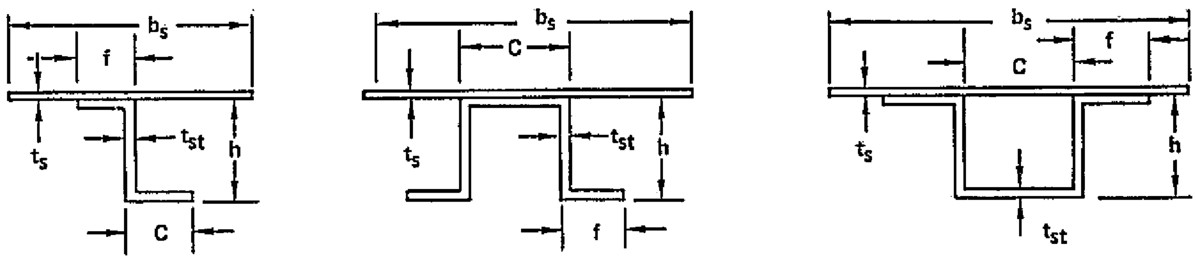
WHERE:

f_n = PRINCIPAL STRESS

$$= \frac{f_x}{2} \pm \left[\left(\frac{f_x}{2}\right)^2 + f_{xy}^2\right]^{1/2}$$

TABLE 12-59. GEOMETRY COMPARISON OF THE CANDIDATE FUSELAGE PANEL CONCEPTS AT SELECTIVE LOCATIONS, TASK I INITIAL SCREENING

POINT DESIGN REGION	LOCATION	PANEL CONCEPT	FUSELAGE PANEL DIMENSION						
			b_s (IN.)	t_s (IN.)	C (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	$\bar{t}$ (IN.)
FS 2000 AND FS 3000	TOP	ZEE STIFF	4.00	.100	0.75	1.00	1.25	0.90	.160
		OPEN HAT	5.00	.080	1.25	.80	1.25	.063	.151
		CLOSED HAT	6.00	.080	1.50	.80	1.25	.063	.142
FS 2500	TOP	ZEE STIFF	4.00	.100	0.75	1.00	1.25	.110	.190
		OPEN HAT	5.00	.080	1.25	.80	1.25	.070	.159
		CLOSED HAT	6.00	.080	1.50	.80	1.25	.080	.159



ZEE-STIFFENED CONCEPT

HAT-STIFFENED (OPEN) CONCEPT

HAT-STIFFENED (CLOSED) CONCEPT

TABLE 12-60. WEIGHT COMPARISON OF THE CANDIDATE FUSELAGE PANEL CONCEPTS, TASK I INITIAL SCREENING

POINT DESIGN REGION	AVERAGE PANEL WEIGHT (B/SQ. FT)		
FS 750	–	–	1.31
FS 2000	2.98	2.80	3.01
FS 2500	3.35	3.18	3.52
FS 3000	2.98	2.80	3.01

NOTES:

1. CONSTANT FRAME SPACING = 20.0 INCHES

components prior to estimating the total fuselage weight, Section 15. Those least weight concepts investigated were the zee-stiffened panel concept at FS 750 and the closed hat-stiffened concept at FS 2000, 2500, and 3000. All designs incorporated the floating frame design with skin shear clips and a minimum-weight frame spacing of 20-inches. In review, the point design locations are presented in Figure 12-4 and the minimum-weight panel and frame concepts are displayed among the list of concepts shown in Figure 12-2.

As with the prior fuselage studies, the shear and bending moment diagrams shown in Figures 12-61 and 12-62 were the basis for defining the point design environment. The panel load intensities used for this analysis were the same theoretical distributions as calculated for the Initial Screening, Table 12-57.

Unlike the previous fuselage analysis, the cabin pressure and thermal environment were included in the definition of the point design environments. Tables 12-62 and 12-63 contain the detail data related to these components. A summary of the point design environment which includes the inplane loads, normal loads (pressure), and thermal components is presented in Table 12-64 for the start-of-cruise design condition.

The fuselage shell was analyzed for its most critical flight condition, the ultimate load condition at start-of-cruise. For this analysis, the biaxial stress state was defined at each point design region by superposing the airload and pressure membrane stresses. The airload membrane forces (N_x and N_{xy}) are contained in the point design environment specified in Table 12-64, and the pressure forces (N_x and N_θ) are defined in Table 12-65. Using these biaxial forces and the initial panel geometry, the biaxial stress state and resulting principal stress are calculated and compared to the applicable allowable stress (tension or compression). This process is repeated until reasonable convergence is attained between the principal and allowable stresses, i.e., positive margin of safety. Table 12-66 presents a summary of the stress levels obtained on the most critical panels at each point design region.

In addition to the membrane analysis conducted on the shell, which is applicable for the shell structure at a reasonable distance from the frame attachment, a discontinuity analysis was conducted at the frame/shell interface to assess the total stress state for both shell and frame. This analysis was performed using the theory presented by Flügge in Reference 7. For this analysis, the membrane stresses due to

TABLE 12-61. WEIGHT COMPARISON OF THE CANDIDATE FUSELAGE ARRANGEMENTS, TASK I INITIAL SCREENING

POINT DESIGN REGION	FRAME UNIT WEIGHT (LB/SQ FT)	FUSELAGE UNIT WEIGHTS (LB/SQ FT)					
		OPEN-HAT		CLOSED-HAT		ZEE-STIFF	
		PANEL	TOTAL	PANEL	TOTAL	PANEL	TOTAL
FS 750	0.25	-	–	–	–	1.31	1.56
FS 2000	0.46	2.98	3.44	2.80	3.26	3.01	3.47
FS 2500	0.60	3.35	3.95	3.18	3.78	3.52	4.12
FS 3000	0.46	2.98	3.44	2.80	3.26	3.01	3.47

TABLE 12-62. FUSELAGE CABIN PRESSURES

CONDITION	WT. X 10^{-3} LBS	MACH NO.	LOAD FACTOR n_z	V_e K_{eas}	ALT. X 10^{-3} FT.	CABIN(1) PRESSURE (PSI)
START-OF-CRUISE	660	2.7	2.5	460	61.5	17.55
TRANSONIC DESCENT AT M1.2	690	1.2	2.5	372	38.2	17.55

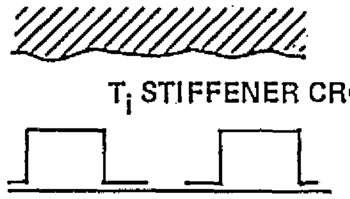
1. ULTIMATE p = 1.5 X LIMIT p

TABLE 12-63. TEMPERATURE AND GRADIENTS FOR FUSELAGE SKIN PANELS - TASK I

NOTES:

1. BASED ON HOT DAY (STD+8K) 4200 n.mi. FLIGHT PROFILE.
2. HAT-STIFFENED PANELS, EXCEPT ZEE-STIFFENED AT FS 750.
3. 'TOP', 'BOTTOM' AT ℄ ; 'SIDE' AT 90° OR ABOVE WING.

PANEL SCHEMATIC



TEMPERATURES IN F

LOCATION	FLIGHT CONDITION			
	START OF CRUISE		MACH 1.2 DESCENT	
	T_i-T_o	T_{AVG}	T_i-T_o	T_{AVG}
TOP				
FS 750	-105	342	+111	114
2000	-175	295	+171	144
2500	-186	281	+181	156
3000	-174	292	+170	145
SIDE				
FS 750	-106	332	+109	108
2000	-157	324	+156	129
2500	-171	311	+170	139
3000	-147	301	+142	122
BOTTOM				
FS 750	-106	333	+109	109
3000	-177	278	+171	141

TABLE 12-64. FUSELAGE POINT DESIGN ENVIRONMENT, DETAILED CONCEPT ANALYSIS

START OF CRUISE; MACH NO. 2.7; n_z=2.5

ITEM	UNITS	FS 750			FS 2000			FS 2500			FS 3000		
		UPPER PANEL	SIDE PANEL	LOWER PANEL	UPPER PANEL	SIDE PANEL	LOWER PANEL	UPPER PANEL	SIDE PANEL	LOWER PANEL	UPPER PANEL	SIDE PANEL	LOWER PANEL
N_x	LB/IN	1580	200	-1580	11630	1230	–	15730	1230	–	11630	1230	-11670
N_{xy}	LB/IN	50	250	50	412	1360	–	629	2025	–	412	1360	415
INTERNAL PRESSURE	PSI	17.55	17.55	17.55	17.55	17.55	–	17.55	17.55	–	17.55	17.55	17.55
T_{AVG}	°F	342	332	333	295	324	–	281	311	–	292	301	278
ΔT	°F	-105	-106	-106	-175	-157	–	-186	-171	–	-174	-147	-177

TABLE 12-65. FUSELAGE SHELL MEMBRANE FORCES DUE TO INTERNAL PRESSURIZATION

POINT DESIGN REGION	R in.	A in.^{2}	C in.	UNIT N_X (lb/in.)	DESIGN (2) PRESSURE p (PSI)	TOTAL N_X (lb/in.)	HOOP N_θ (lb/in.)
750	72.0	11,761	411	28.6	17.55	502	1264
2000	68.0	10,787	394	27.4	17.55	480	1193
2500	68.0	10,787	394	27.4	17.55	480	1193
3000	61.0	11,690	383	30.5	17.55	535	1070

1. NOMENCLATURE

   R = SHELL RADIUS, in.
   A = ENCLOSED PRESSURIZED AREA, in.^{2}
   C = SHELL CIRCUMFERENCE, in.
   UNIT N_X = A/C, lb/in. per psi
   TOTAL N_X = px(UNIT N_X); lb/in.
   N_θ = p x R

2. ULTIMATE DESIGN PRESSURE FOR START-OF-CRUISE FLIGHT CONDITION

3. PRESSURIZED REGION

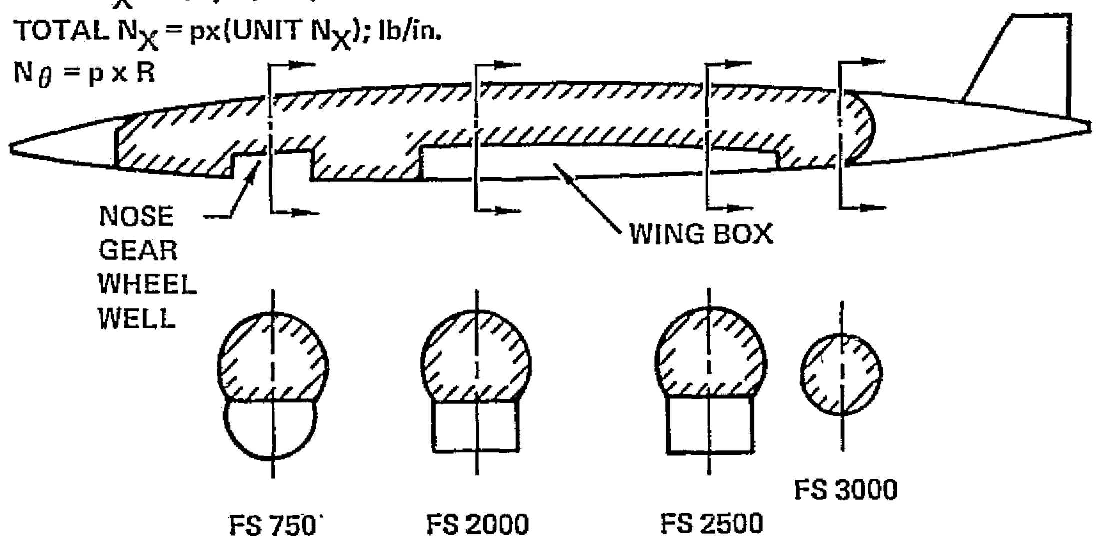


TABLE 12-66. SUMMARY OF FUSELAGE SHELL STRESS LEVELS

POINT DESIGN REGION	PANEL LOCATION	LOAD INTENSITY (ULT.), LB./IN.(1)					PANEL GEOMETRY(2)		STRESS LEVEL (ULT.), PSI(3)			
		AXIAL LOAD, N_x			HOOP LOAD N_θ	SHEAR FLOW N_{xy}						
		AIR LOAD	PRESS. LOAD	TOTAL LOAD			$\bar{t}$ IN.^{2}/IN.	t_s IN.	f_x	f_θ	f_{xy}	f_n
FS 750	UPPER	1,580	502	2,082	1,264	50	0.056	0.036	37,200	35,100	1,400	37,900
	SIDE	200	502	702	1,264	250	0.056	0.036	12,500	35,100	6,900	37,000
	LOWER	-1,580	502	-1,078	1,264	50	0.056	0.036	-19,300	35,100	1,400	35,100
FS 2000	UPPER	11,600	480	12,080	1,193	412	0.145	0.080	83,300	14,900	5,200	83,700
	SIDE	377	480	857	1,193	1,361	0.099	0.063	8,700	18,900	21,600	36,000
FS 2500	UPPER	15,700	480	16,180	1,193	629	0.184	0.100	87,900	11,900	6,300	88,400
	SIDE	422	480	902	1,193	2,025	0.109	0.063	8,300	18,900	32,100	45,800
FS 3000	UPPER	11,600	535	12,135	1,070	412	0.145	0.080	83,700	13,400	5,200	84,100
	SIDE	377	535	912	1,070	1,361	0.099	0.063	9,200	17,000	21,600	35,000
	LOWER	-11,700	535	-11,165	1,070	415	0.177	0.090	-63,100	11,900	4,600	-63,400

1. LOAD INTENSITIES PER POINT DESIGN ENVIRONMENT AND PRESSURE FORCE TABLES

2. PANEL GEOMETRY:

   $\bar{t}$ = EQUIVALENT PANEL THICKNESS

   t_s = SKIN THICKNESS

3. STRESS LEVEL:

$$f_x = N_x(\text{TOTAL})/\bar{t}$$

$$f_\theta = N_\theta / t_s$$

$$f_{xy} = N_{xy}/t_s$$

$$f_n = \frac{f_x + f_\theta}{2} \pm \left[\left(\frac{f_x - f_\theta}{2}\right)^2 + f_{xy}^{\,2}\right]^{1/2}$$

4. SIGN CONVENTION:

   POSITIVE = TENSION

   NEGATIVE = COMPRESSION

the airload and internal pressure are superposed upon the discontinuity stresses (bending and shear stresses) caused by the pressure and thermal gradients between the shell and frame. Typical results of this analysis are shown in Figures 12-66 and 12-67. These figures display the shell hoop stresses and the stringer bending moment caused by the pressure and thermal environment during the operating condition (mid-cruise, limit one-g condition). The hoop stresses shown in Figure 12-66 are compared to the operating fatigue allowable of 25,000 psi.

As a result of the preceding analyses, the fuselage shell and frames were sized for their critical failure mode at each point design region. Table 12-67 shows the fuselage shell geometry at the most critical circumferential locations for each region; whereas, Table 12-68 displays the circumferential variation in the panel geometry at one point design region, FS 2500. Similarly, the frame equivalent thicknesses are shown in Table 12-69 and includes the component (frame and shear tie) and total thicknesses requirements at specific circumferential locations, and the average equivalent panel thickness for the frame at each point design region.

In conclusion, Table 12-70 summarizes the component thicknesses, total equivalent thickness for each point design region, and the corresponding unit weights. A maximum weight of 3.53 lb/sq.ft. is indicated for the maximum fuselage bending region at FS 2500; whereas, the corresponding panel at the lightly loaded forebody region (FS 750) has a weight that is approximately 45-percent lighter, i.e., 1.54 lb/sq.ft. Unit weights of 3.27 lb/sq.ft. and 3.43 lb/sq.ft. are indicated for FS 2000 and FS 3000, respectively. These unit weights are used as the basis for predicting the total weight of the fuselage. The Mass Section of this report (Section 15) describes the methods and results obtained from extrapolating these weights to total fuselage weight.

CHORDWISE STIFFENED WING ARRANGEMENT - TASK IIA

Modification of the Task I Baseline airplane was required prior to commencing the Task II detailed engineering studies. These modifications encompassed shortening the fuselage forebody, changing the sweep-angle on the wing tip leading edge and relocating some of the fuel tanks. A more detailed description of these changes are presented in Section 2.

C.3

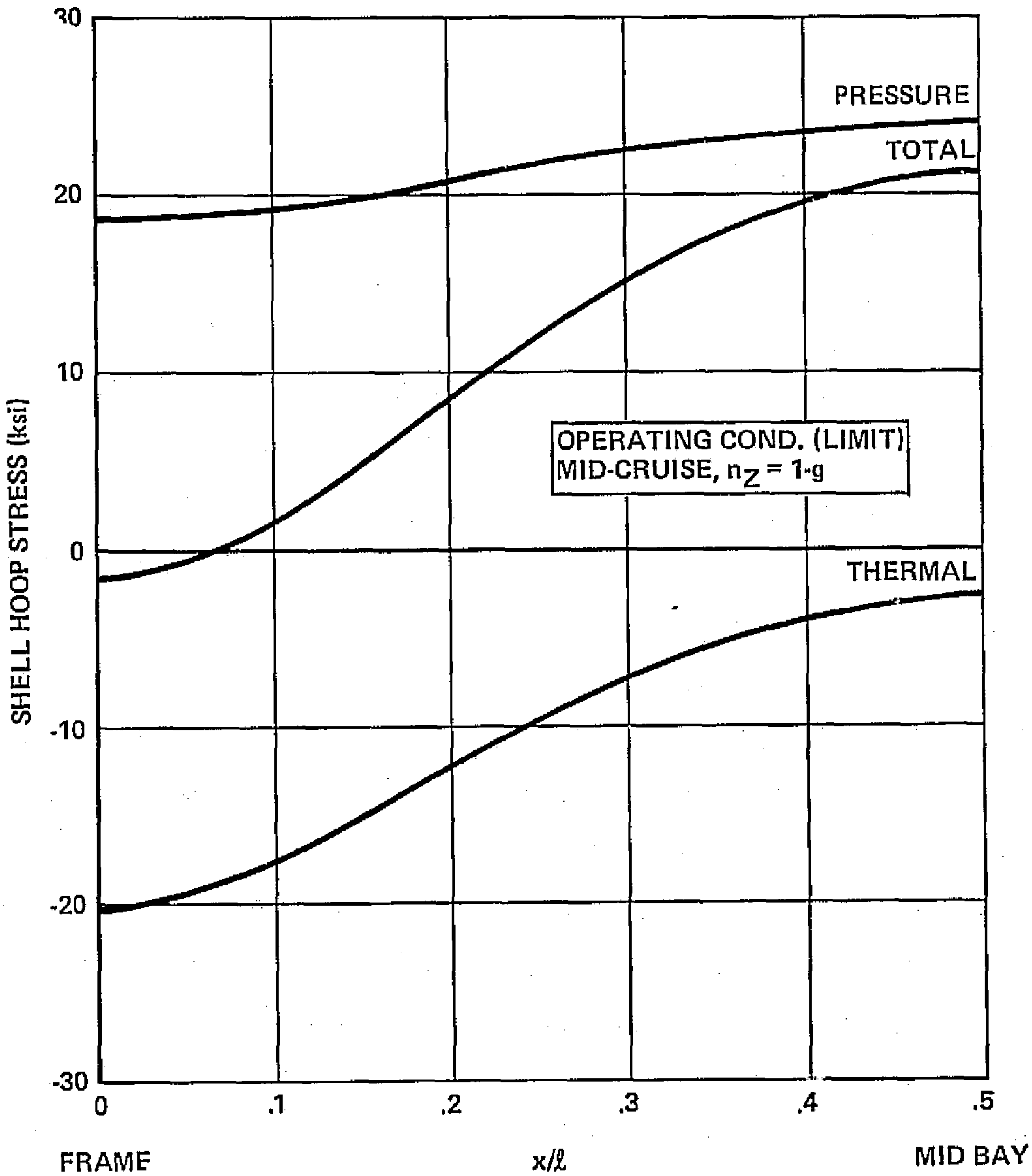


Figure 12-66. Shell Hoop Stresses due to Discontinuity Forces, Point Design Region FS 750

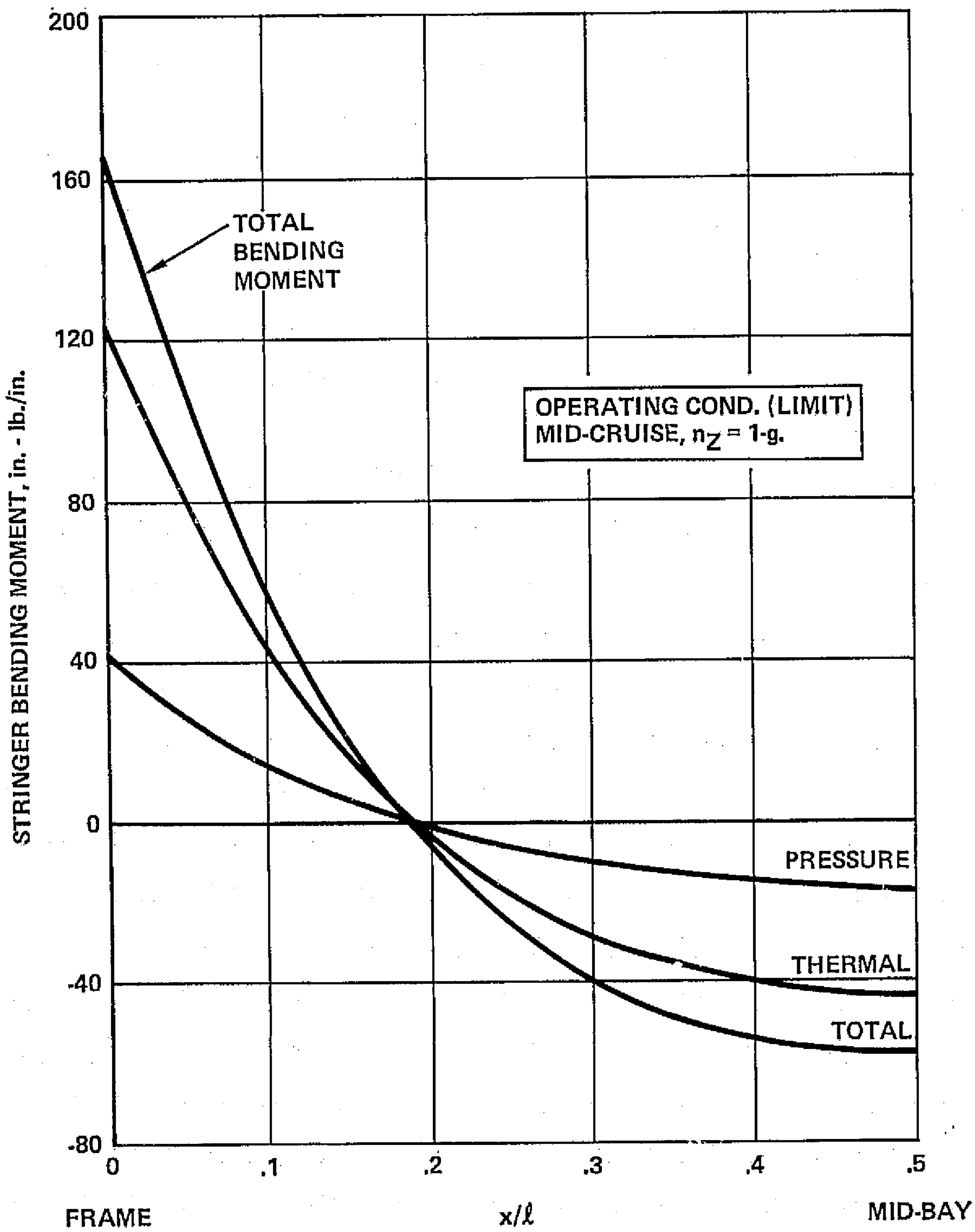
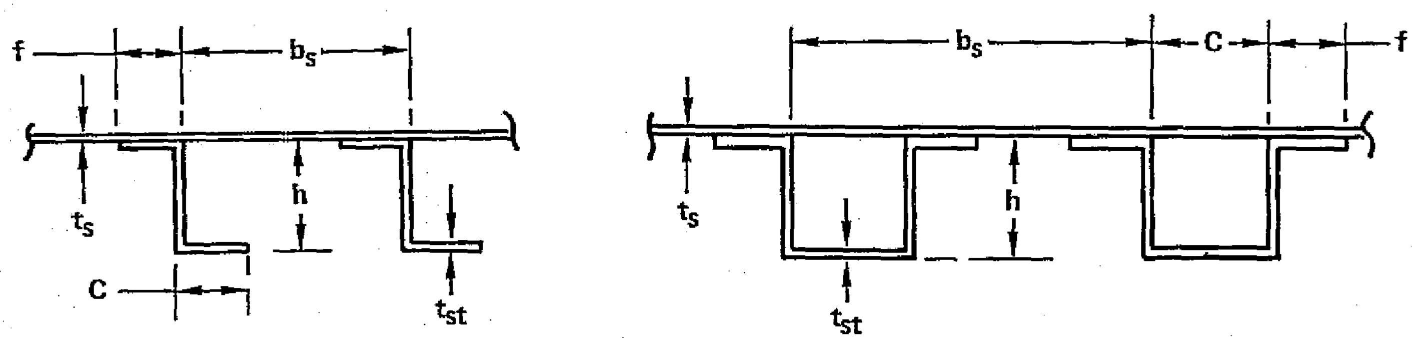


Figure 12-67. Stringer Bending Moments due to Discontinuity Forces, Point Design Region FS 750

TABLE 12-67. COMPARISON OF FUSELAGE PANEL GEOMETRY – DETAILED CONCEPT ANALYSIS

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	FUSELAGE PANEL DIMENSION b_s (IN.)	t_s (IN.)	C (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	t (IN.)
FS 750	ZEE-STIFFENED	TOP	4.0	.036	.55	.75	1.00	.036	.056
		SIDE	4.0	.036	.55	.75	1.00	.036	.056
		BOTTOM	4.0	.036	.55	.75	1.00	.036	.056
FS 2000	HAT-STIFFENED	TOP	6.0	.080	1.5	.80	1.25	.070	.145
		SIDE	6.0	.063	1.5	.75	1.25	.040	.099
FS 2500	HAT-STIFFENED	TOP	6.0	.100	1.5	.80	1.25	.090	.184
		SIDE	6.0	.063	1.5	.75	1.25	.050	.109
FS 3000	HAT-STIFFENED	TOP	6.0	.080	1.5	.80	1.25	.070	.145
		SIDE	6.0	.063	1.5	.75	1.25	.040	.099
		BOTTOM	6.0	.090	1.5	.90	1.25	.090	.177

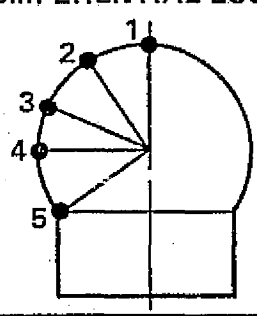


ZEE-STIFFENED CONCEPT HAT-STIFFENED CONCEPT

TABLE 12-68. FUSELAGE PANEL GEOMETRY AT FS 2500,
DETAILED CONCEPT ANALYSIS

POINT DESIGN REGION	PANEL CONCEPT	CIRCUM. LOCATION	FUSELAGE PANEL DIMENSIONS						
			b_s (IN)	t_s (IN)	C (IN)	f (IN)	h (IN)	t_{st} (IN)	$\bar{t}$ (IN)
FS 2500	HAT-STIFFENED	1 (TOP)	6.0	.100	1.50	.80	1.25	.090	0.184
		2	6.0	.070	1.50	.75	1.25	.070	0.134
		3	6.0	.063	1.50	.75	1.25	.063	0.121
		4 (SIDE)	6.0	.063	1.50	.75	1.25	.050	0.109
		5	6.0	.070	1.50	.75	1.25	.063	0.128

CIRCUMFERENTIAL LOCATIONS:



PANEL DIMENSIONS:

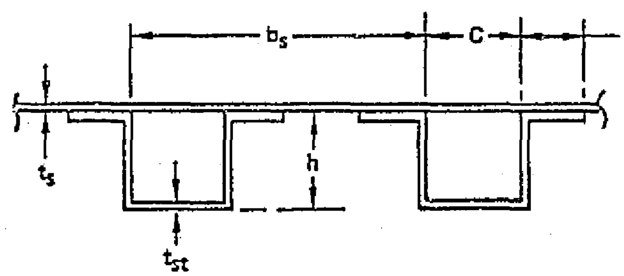


TABLE 12-69. FUSELAGE FRAME WEIGHTS, DETAILED CONCEPT ANALYSIS

POINT DESIGN REGION	CIRCUM. LOCATION	FRAME SPACING (IN.)	EQUIVALENT PANEL THICKNESS ($\bar{t}$), IN.^{2}/IN.			
			FRAME	SHEAR TIE	TOTAL	AVERAGE
FS 750	ALL	20.0	.007	.004	.011	(.011)
FS 2000	UPPER	20.0	.018	.006	.024	(.023)
	SIDE	20.0	.016	.006	.022	
FS 2500	UPPER	20.0	.016	.006	.022	(.022)
	SIDE	20.0	.016	.006	.022	
FS 3000	UPPER	20.0	.018	.006	.024	(.023)
	SIDE	20.0	.015	.006	.021	
	LOWER	20.0	.019	.007	.026	

$\bar{t}$(TOTAL) = $\bar{t}$(FRAME + $\bar{t}$(SHEAR TIE)

FRAME GEOMETRY

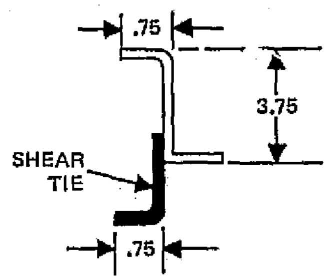


TABLE 12-70. FUSELAGE WEIGHT SUMMARY, DETAILED CONCEPT ANALYSIS

POINT DESIGN REGION	PANEL CONCEPT	EQUIV. PANEL THICKNESS (IN.^{2}/IN.)			UNIT WEIGHT W (LB/SQ. FT)
		FRAME $\bar{t}$	PANEL $\bar{t}$	TOTAL $\bar{t}$	
FS 750	ZEE-STIFF.	0.011	0.056	0.067	1.54
FS 2000	HAT-STIFF.	0.023	0.119	0.142	3.27
FS 2500	HAT-STIFF.	0.022	0.131	0.153	3.53
FS 3000	HAT-STIFF.	0.023	0.126	0.149	3.43

To assess the results of these modifications on structural weight, an investigation was conducted using the Task I chordwise finite-element 2-D model and included:

- Obtaining the structural influence coefficients (SIC) with the revised fuselage and wing tip. The wing and fuselage flexibilities (section properties) were held constant for this solution.
- Modifying the net aeroelastic loads for a critical Task I load condition to reflect the changes in the aerodynamic and inertia load components.
- Conducting a design loads run to obtain new internal loads using the stiffness of the modified structural model and the revised net aeroelastic loads.
- Performing a weight-strength analysis on the major wing structural components at selective locations using the revised load intensities and comparing these results with those of the Task I analysis.

Point Design Environment

As the basis for the structural analysis the point design environment was defined for a critical flight condition at several wing point design regions. The regions selected for analysis were the forward wing box region 40322 and the aft wing box region 40536. The flight condition selected was the flutter critical Mach 0.9 subsonic flight condition and included the following load factors; positive 1.0-g, a positive 2.5-g steady-state maneuver, a positive 2.5-g transient maneuver, and a negative 1.0-g flight attitude.

Using this load condition, the in-plane loads were determined by performing a NASTRAN static solution with the modified 2-D structural model. A summary of these wing surface load intensities results are presented in Table 12-71 along with the results of the Task I analysis for comparison purposes.

The point design environment at regions 40322 and 40536 were defined using the new surface panel load intensities resulting from the NASTRAN solution, with the same pressure and temperature components derived during the Task I analysis. A comparison of the Task I and Task IIA point design environment at regions 40322 and 40536 are shown in Tables 12-72 and 12-73. The critical Task I and Task IIA design conditions are presented in the footnotes of these tables.

TABLE 12-71. COMPARISON OF TASK I AND TASK IIA WING LOAD INTENSITIES, MACH 0.90 LOAD CONDITION

PANEL IDENTIFICATION			*LOAD INTENSITY (ULTIMATE), LBS/IN.			
			TASK I			IIA
REGION	NUMBER	DIRECTION	CHORDWISE	SPANWISE	MONOCOQUE	CHORDWISE
WING-FORWARD	40322	Nx	– 10	– 148	– 199	– 819
		Ny	– 1145	– 1155	– 595	– 1120
		Nxy	201	275	211	143
WING-AFT BOX	40236	Nx	188	122	– 925	– 377
		Ny	–10846	–12181	–8102	–11474
		Nxy	418	1181	858	436
	40536	Nx	85	– 132	–1483	– 471
		Ny	–10680	–12318	–8763	–11207
		Nxy	1118	2288	2521	1409
	41036	Nx	– 274	– 36	–1094	– 567
		Ny	– 6570	– 6876	–4544	– 7040
		Nxy	1369	2027	1949	1581
WING-TIP	41316	Nx	701	298	– 932	592
		Ny	–11655	–12546	–8268	–12145
		Nxy	3492	3240	2528	3773
	41348	Nx	– 719	– 574	– 605	– 1068
		Ny	– 6293	– 5886	–4731	– 6402
		Nxy	1535	1797	2132	1990

*LOAD CONDITIONS:
TASK I CONDITION 12: MACH 0.90, nz = 2.5, W = 700,000 LB, Ve = 325 KEAS
TASK IIA CONDITION 9: MACH 0.90, nz = 2.5, W = 700,000 LB, Ve - 325 KEAS

TABLE 12-72. TASK IIA WING POINT DESIGN ENVIRONMENT, MACH 0.90 LOAD CONDITIONS

SYMMETRICAL FLIGHT, STEADY MANEUVER AT MACH 0.90 (V_c)

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION			
			40322		40536	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	Nx	LB/IN	-1207	1207	-471	471
	Ny	LB/IN	-1081	1081	-11207	11207
	Nxy	LB/IN	176	176	1409	1409
THERMAL STRAIN	ϵx	IN/IN	–	–	–	–
	ϵy	IN/IN	–	–	–	–
	ϵxy	IN/IN	–	–	–	–
PRESSURE	AERO	PSI	-1.40	-0.30	-1.35	-0.49
	FUEL	PSI	-6.74	-8.96	-5.63	-7.91
	NET	PSI	-8.14	-9.26	-6.98	-8.40
TEMPERATURE	T_{AV}	°F	50	53	52	54
	ΔT	°F	132	38	29	32

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

(3) DESIGN CONDITIONS: REGION 40322 - COND. 10, REGION 40536 - COND. 9

TABLE 12-73. TASK I WING POINT DESIGN ENVIRONMENT, MACH 0.90 LOAD CONDITION

SYMMETRICAL FLIGHT, STEADY MANEUVER AT MACH 0.9 (V_c)

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION			
			40322		40536	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
IAR LOADS	Nx	LB/IN	-531	531	85	-271
	Ny	LB/IN	-1115	1115	10680	9777
	Nxy	LB/IN	-236	236	1118	778
THERMAL STRAIN	ϵx	IN/IN	–	–	–	–
	ϵy	IN/IN	–	–	–	–
	ϵxy	IN/IN	–	–	–	–
PRESSURE	AERO	PSI	-1.40	-0.30	-1.35	-0.49
	FUEL	PSI	-6.74	-8.96	-5.63	-7.91
	NET	PSI	-8.14	-9.26	-6.98	-8.40
TEMPERATURE	T_{AV}	°F	50	53	52	54
	ΔT	°F	132	38	29	32

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOADS SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

(3) DESIGN CONDITIONS: REGION 40322 AND LOWER SURFACE AT 40536 - COND. 13, UPPER SURFACE AT REGION 40536 - COND. 12

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Weight/Strength Analysis

To assess the results of the airplane configuration modification on structural weight, the major weight components of the chordwise wing arrangement were analyzed for the modified Task IIA point design environments. For comparison purposes, the same components were reanalyzed using the Task I load/temperature environment corresponding to the single flight condition investigated for Task IIA. These components included the upper and lower surface panels (circular arc convex beaded concept) and spar caps. A comparison of the Task I and Task IIA panel results at regions 40322 and 40536 are shown in Table 12-74 and includes the panel cross-sectional dimensions and weight data for the 20-inch spar spacing designs. In general for these two regions, the Task IIA panel designs are heavier than the corresponding Task I designs with a maximum weight increase of 23-percent noted for the Task IIA upper surface panel at region 40322. The exception being the Task IIA lower surface panel at region 40322 which is approximately 3-percent lighter than the Task I design.

A comparison of the panel weight/strength analysis results are presented in Table 12-75 and includes the unit weights for the surface panels and the panel relative weight factor, i.e., weight of the Task IIA panel divided by the weight of the Task I panel.

For the chordwise wing arrangement the spar caps are major weight components, these spar caps are uniaxially loaded by the spanwise bending loads; hence, the cap weights are directly proportional to the spanwise surface load intensities. Table 12-76 contains a comparison of the spanwise load intensities (N_y) for the two point design regions and the relative weight factor of the Task IIA spar caps.

The results of this analysis reflect the strength-sizing of the major chordwise wing components for a flutter critical flight condition (Mach .9 symmetric flight condition). Since the resulting panel and spar cap weights do not necessarily reflect the most critical static strength condition, the relative weight of the components were used for comparing unit box weights.

A comparison of the Task I and Task IIA unit box weights for regions 40322 and 40536 are shown in Table 12-77. The Task I values reflect the results of the previously conducted analysis on the chordwise wing arrangement (sized for the most critical flight condition); whereas, the Task IIA values reflect the Task I weights multiplied by the relative weight factors presented in Tables 12-75 and 12-76. With

TABLE 12-74. COMPARISON OF TASK I AND TASK IIA WING PANEL GEOMETRY AND WEIGHT

POINT DESIGN REGION	40322				40536			
TASK	I		IIA		I		IIA	
SURFACE	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPAR SPACING (IN.)	20	20	20	20	20	20	20	20
DIMENSIONS:								
t_ℓ (IN.)	.015	.013	.018	.012	.018	.017	.020	019
t_u (IN.)	.015	.020	.019	.020	.020	.020	.024	.020
R_ℓ (IN.)	.900	.800	1.000	1.000	.800	.800	.800	.700
θ (DEG.)	87.000	87.000	87.000	87.000	87.000	87.000	87.000	87.000
b (IN.)	.750	.750	.750	.750	.750	.750	.750	.750
MASS DATA:								
$\bar{t}$ (IN.)	.036	.038	.044	.037	.045	.043	.052	.046
w (LB./FT.^{2})	.825	.875	1.018	.851	1.031	1.000	1.187	1.055
CRITICAL CONDITION	13	13	10	10	12	13	9	9

PANEL CONCEPT:

CIRCULAR ARC – CONVEX BEADED SKIN (h/c = 0.10)

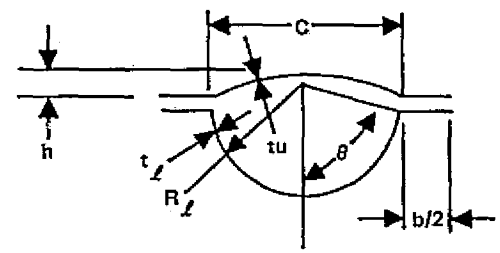


TABLE 12-75. COMPARISON OF TASK I AND TASK IIA
SURFACE PANEL WEIGHTS

TASK NO.		I	IIA	RELATIVE WEIGHT FACTOR
POINT DESIGN REGION	SURFACE	PANEL WEIGHT (LB/SQ. FT)		
40322	UPPER	0.825	1.018	1.23
	LOWER	0.875	0.851	0.97
40536	UPPER	1.031	1.187	1.15
	LOWER	1.000	1.055	1.06

TABLE 12-76. COMPARISON OF TASK I AND TASK IIA
SPAR CAP LOADS

TASK NO.		I	IIA	RELATIVE WEIGHT FACTOR
POINT DESIGN REGION	SURFACE	SPANWISE LOAD INTENSITY (LB/IN.)		
40322	UPPER	-1,115	-1,081	0.97
	LOWER	1,115	1,081	0.97
40536	UPPER	-10,680	-11,207	1.05
	LOWER	10,680	11,207	1.05

TABLE 12-77. COMPARISON OF THE DETAIL WING WEIGHTS FOR THE TASK I AND TASK IIA CHORDWISE STIFFENED WING ARRANGEMENTS

POINT DESIGN REGION	40322			40536		
TASK NO.	I	IIA	RELATIVE WEIGHT	I	IIA	RELATIVE WEIGHT
SPAR SPAC (IN.)	20	20		20	20	
PANELS						
UPPER	0.825	1.015	1.23	1.609	1.850	1.15
LOWER	0.942	0.914	0.97	1.335	1.415	1.06
Σ	(1.767)	(1.929)	(1.09)	(2.944)	(3.265)	(1.11)
RIB WEBS						
BULKHEAD	0.298	0.298	1.00	0.238	0.238	1.00
TRUSS	0.074	0.074	1.00	0.228	0.228	1.00
Σ	(0.372)	(0.372)	(1.00)	(0.466)	(0.466)	(1.00)
SPAR WEBS						
BULKHEAD	0.336	0.336	1.00	0.270	0.270	1.00
TRUSS	0.301	0.301	1.00	0.490	0.490	1.00
Σ	(0.637)	(0.637)	(1.00)	(0.760)	(0.760)	(1.00)
RIB CAPS						
UPPER	0.058	0.058	1.00	0.116	0.116	1.00
LOWER	0.065	0.065	1.00	0.086	0.086	1.00
Σ	(0.123)	(0.123)	(1.00)	(0.202)	(0.203)	(1.00)
SPAR CAPS						
UPPER	0.241	0.234	0.97	2.710	2.846	1.05
LOWER	0.350	0.340	0.97	3.950	4.148	1.05
Σ	(0.591)	(0.574)	(0.97)	(6.660)	(6.994)	(1.05)
NON-OPTIMUM						
MECH. FAST.	0.180	0.180	1.00	0.200	0.200	1.00
WEB INTERS.	0.120	0.120	1.00	0.120	0.120	1.00
Σ	(0.300)	(0.300)	(1.00)	(0.320)	(0.320)	(1.00)
Σ POINT DESIGN MASS $\frac{LB}{FT^2}$	3.790	3.935	1.04	11.352	12.007	1.06

respect to Table 12-77, the detail weight statements includes the surface panels, rib webs, spar webs, rib caps, spar caps, and the non-optimum weights. Only the weights of the Task IIA panels and spar caps were altered, the remaining components were taken directly from the Task I analysis. In addition to the components weight, the relative weight factors are shown. Region 40322 indicates the Task IIA configuration has heavier panels (approximately 9-percent), lighter spar caps (approximately 3-percent), and a unit weight that is 4-percent heavier. For region 40536, the Task IIA panels are 11 percent heavier, spar caps are 5-percent heavier, and the unit weight is 6 percent heavier.

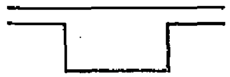
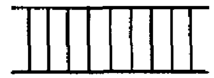
WING STRUCTURAL ARRANGEMENT - TASK IIB

The wing structural arrangement selected for evaluation in the Task II Detailed Engineering Study was comprised of the most promising structural-material concepts surviving the Task I Analytical Design Studies. In review, the Task I'analysis resulted in the selection of the least-weight arrangements from each basic type of wing load-carrying structure (chordwise, spanwise, and monocoque). Those five arrangements selected for the final Task I detail evaluation with respect to weight, cost, performance, and risk included:

- The two chordwise arrangements corresponding to the least-weight metallic and composite reinforced designs. Both designs incorporated the least-weight metallic panel concept (circular-arc convex-beaded design). For the substructure the metallic design employs all titanium alloy spar caps, and the composite reinforced design employs a titanium alloy spar cap reinforced with unidirectional Boron/polyimide (B/PI).
- The least-weight spanwise arrangement, metallic hat-stiffened skin panels with representative substructure.
- The two monocoque arrangements, which are characterized by their respective panel-to-substructure attachment design, tubular insert-welded and densified core - mechanically fastened. Both concepts incorporated aluminum brazed honeycomb-core sandwich panels.

The complete results of this detail evaluation are described in the section entitled Concept Evaluation and Selection, Section 17. For summary purposes, the results of the weight evaluation are repeated in Table 12-78 which lists the variable and fixed weights for each arrangement. Furthermore, the variable weight is defined for the

TABLE 12-78. WING WEIGHTS FOR STRUCTURAL ARRANGEMENTS

WING WEIGHT AND SEGMENT	STRUCTURAL ARRANGEMENT				
	CHORDWISE	SPANWISE	MONOCOQUE	MONOCOQUE	CHORDWISE
	WELD BOND  MECH. FASTEN.	WELD BOND  MECH. FASTEN.	ALUM BRAZED  MECH. FASTEN.	ALUM BRAZED  WELDED	COMP. REINF.  SPARS ONLY
VARIABLE WEIGHT	64,658	63,482	50,978	53,794	48,082
● FWD. BOX	(22,090)	(25,364)	(21,982)	(24,057)	(20,580)
● AFT BOX	(29,016)	(25,242)	(19,692)	(20,153)	(17,384)
● TIP	(13,552)	(12,876)	(9,304)	(9,584)	(10,118)
FIXED WEIGHT	41,352	41,352	41,352	41,352	41,352
Σ TOTAL~LB	106,010	104,834	92,330	95,146	89,434

major wing areas, i.e., forward box, aft box, and wing tip. From a review of these variable weights, it can be seen that a structural arrangement composed of the lowest-weight designs for each area would afford the minimum-weight overall design. Based on this premise, the structural approach selected for further evaluation in the Task II Detail Engineering Studies was a hybrid structural arrangement consisting of a combination of the chordwise-stiffened and monocoque arrangements as shown in Figure 12-68.

Point Design Environment

Similar to the Task I analyses, the hybrid wing structural arrangement was subjected to point design analysis at discrete wing locations (regions). These wing locations, point design regions, are shown in Figure 12-3. The structural definition at each of these regions is in agreement with the combination of structural concepts included in the overall wing structural arrangement, with the load/temperature environment based on the internal load resulting from the NASTRAN static solution using a 3-D finite-element model.

These point design environments were defined for the hybrid wing structural arrangement for both the strength design and strength/stiffness design phases. Examples of these environments are contained in the following text within the discussion for each specific design phase.

Strength Design

The strength design airframe, as characterized by the element properties contained in the finite-element structural model, was developed for the hybrid wing structural arrangement using the results of the Task I analysis. More specifically, the section properties resulting for the Task I strength analysis conducted on the selected chordwise and monocoque concepts were combined to define the total wing stiffness for the finite element model.

Using this strength design model, an internal load solution was obtained using the static aeroelastic loads. These internal loads, in combination with the pressure and temperature components defined in Task I, were used to define the specific point design environments for the structural analysis. Table 12-79 contains the wing point design environment for the symmetrical flight conditions at Mach 0.90 and Mach 1.25.

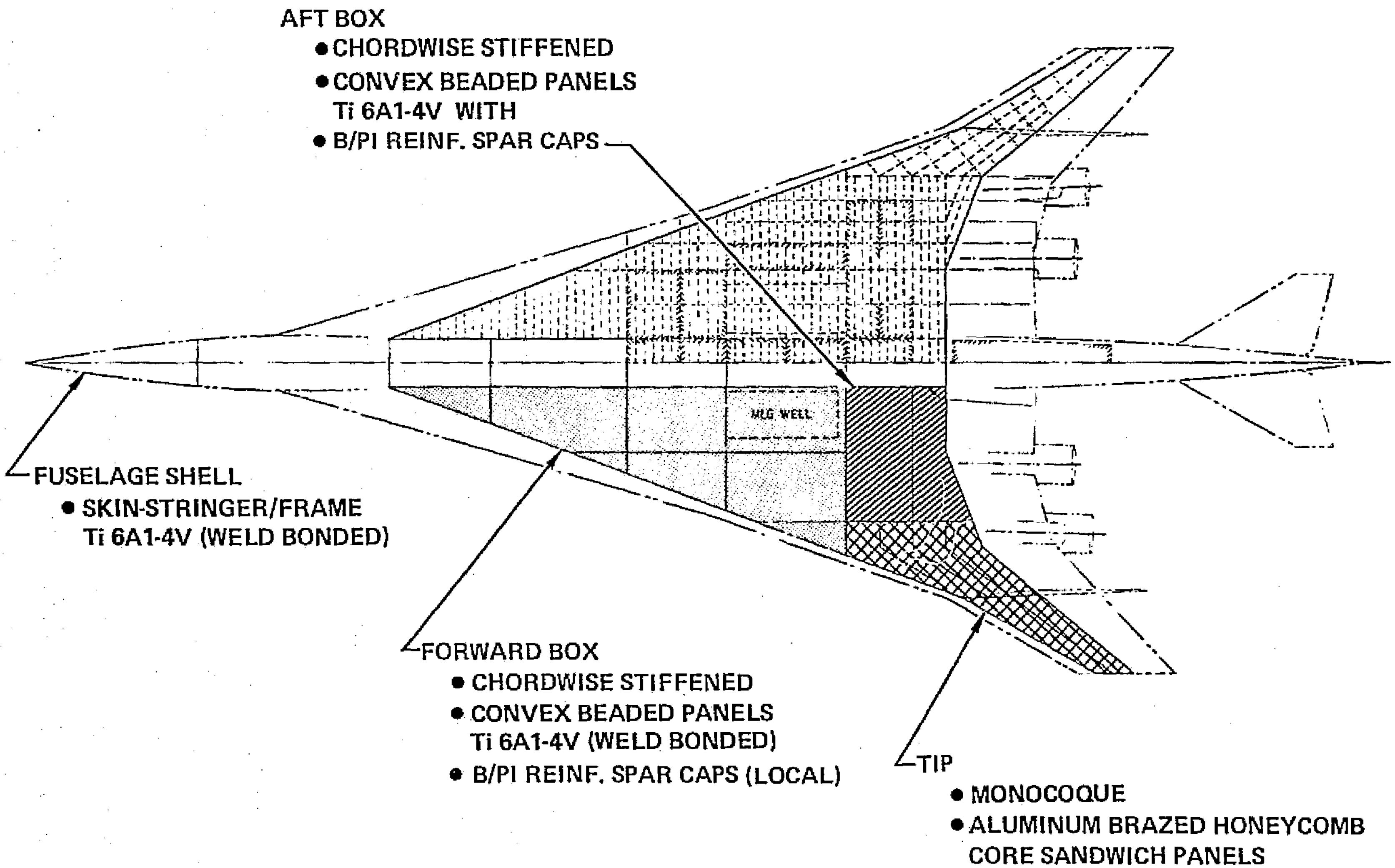


Figure 12-68. Structural Approach for Task II

TABLE 12-79. WING POINT DESIGN ENVIRONMENT, STRENGTH DESIGN - TASK IIB, MACH 0.90 (V_C) AND 1.25 (V_S) LOAD CONDITIONS

CONDITION (8) WEIGHT = 700,000 LB.; MACH 0.90; h = 30,000 FT.; V_e = 325 keas

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION											
			40236		40536		41036		40322		41316		41348	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	N_x	LB/IN	+ 179	+ 750	- 458	+ 582	-1,052	+ 606	- 122	+ 649	-1,226	-1,004	- 877	+ 989
	N_y	LB/IN	-12,779	+13,568	-12,680	12,871	-3,522	+3,074	-1,109	+1,350	-9,504	-8,546	-5,148	+4,867
	N_{xy}	LB/IN	- 271	- 32	- 1,068	+ 1,256	+1,583	+1,542	+ 112	+ 233	+3,686	-3,062	+2,290	-2,414
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	PSI	- 0.99	- 0.44	- 1.35	- 0.49	- 2.10	- 0.36	- 1.40	- 0.30	- 6.75	- 3.76	- 3.68	- 0.30
	FUEL	PSI	- 5.01	- 9.24	- 4.50	- 4.50	0	0	- 4.91	- 8.70	0	0	0	0
	NET	PSI	- 6.00	- 9.68	- 5.85	- 4.99	- 2.10	- 0.36	- 6.31	- 9.00	- 6.75	- 3.76	- 3.68	- 0.30
TEMPERATURE	T_{AV}	°F	47	53	45	45	48	41	47	52	48	41	44	33
	ΔT	°F	+ 37	+ 40	+ 30	+ 36	+ 27	+ 15	+ 37	+ 41	+ 27	+ 15	+ 20	+ 8

CONDITION (12) WEIGHT = 690,000 LB.; MACH 1.25; h = 43,000 FT.; V_e = 294.3 keas

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION											
			40236		40536		41036		40322		41316		41348	
			UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE	UPPER SURFACE	LOWER SURFACE
AIR LOADS	N_x	LB/IN	- 67	+ 246	- 1,073	+ 1,099	-1,812	+1,297	- 151	+ 597	- 1,638	- 1,405	-1,207	+1,379
	N_y	LB/IN	-14,650	+15,197	-14,303	+14,014	-4,220	+3,587	-1,106	+1,400	-12,407	+11,188	-6,897	+6,657
	N_{xy}	LB/IN	- 453	+ 367	- 1,495	+ 1,599	-2,106	+1,909	+ 130	+ 215	+ 4,009	- 2,990	+2,284	-2,281
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	PSI	- 3.03	- 1.20	- 1.27	- .26	- 1.27	- .11	- 1.47	- .06	- 4.98	- .26	- 5.07	- .96
	FUEL	PSI	- 5.08	- 9.31	- 4.50	- 4.50	0	0	- 4.97	- 8.89	0	0	0	0
	NET	PSI	- 8.11	- 10.51	- 5.77	- 4.76	- 1.27	- .11	- 6.44	- 8.83	- 4.98	- .26	- 5.07	- .96
TEMPERATURE	T_{AV}	°F	0	0	0	0	0	0	0	0	0	0	0	0
	ΔT	°F	0	0	0	0	0	0	0	0	0	0	0	0

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

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A strength sizing analysis was conducted on both panels and substructure to assess the weight trends and verify the airframe stiffnesses contained in structural model. While this analysis was being conducted, the vibration and flutter analysis was initiated using the mass and stiffness matrices associated with the strength design airframe. The results of these analyses (flutter and strength) are combined to develop the strength/stiffness airframe design which is discussed in the next section. The results of the strength analysis are discussed in the following text.

Panel Analysis - The panel concepts and their applicable point design regions are shown in Table 12-80 for the hybrid arrangement. These panels were analyzed using the computerized methods previously discussed in Task I and the new point design environment based on the NASTRAN static solution using the 3-D structural model. The results of this strength analysis are shown in Tables 12-81 and 12-82 for the chordwise and monocoque panel concepts, respectively.

With reference to the chordwise panels, circular-arc convex beaded concept, Table 12-81 indicates the foreign object damage (F.O.D.) constraint on the exposed surface bead was active for each point design region with the exception of upper surface panel at Region 40536 ($t_u < 0.015$). The unit weights ranged from a minimum weight of 0.76 lb/sq.ft for the upper surface panel at Region 40322 to a maximum weight of 1.34 lb/sq.ft for the upper surface panel at Region 40536.

The results of the strength analysis conducted on the honeycomb sandwich panels are shown in Table 12-82 for Regions 41036, 41316, and 41348. No thickness constraints were active for the face sheets of these designs which indicated a minimum-weight of 1.20 lb/sq.ft occurring at Region 41036 for both upper and lower surface panels.

TABLE 12-80. SURFACE PANEL CONCEPTS FOR TASK II

STRUCTURAL ARRANGEMENT	POINT DESIGN REGIONS	PANEL CONCEPT	GEOMETRY
CHORDWISE STIFFENED	40322, 40236, 40536	CIRCULAR-ARC/CONVEX BEADED SKIN	
MONOCOQUE	41036, 41316, 41348	HONEYCOMB SANDWICH – ALUMINUM BRAZED	

TABLE 12-81. PANEL GEOMETRY AND WEIGHT FOR THE CHORDWISE STIFFENED PANELS - TASK IIB

DESIGN DATA	POINT DESIGN REGIONS					
	40322		40236		40536	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
RIB	60.0	60.0	60.0	60.0	60.0	60.0
SPAR	22.7	22.7	21.2	21.2	21.2	21.2
DIMENSIONS						
t_L, in.	.013	.015	.015	.020	.023	.019
t_u, in.	.015	.020	.015	.020	.026	.020
R_L, in.	.80	1.00	.80	1.00	.90	.70
θ, degrees	87	87	87	87	87	87
b, in.	.75	.75	.75	.75	.75	.75
pitch, in.	2.35	2.75	2.35	2.75	2.55	2.15
WEIGHT DATA						
$\bar{t}$, in.	.033	.041	.036	.048	.058	.046
W, lb./sq.ft.	.760	.945	.829	1.11	1.34	1.05
CRITICAL DESIGN COND.	12	20	16	16	12	12

DIMENSIONS:

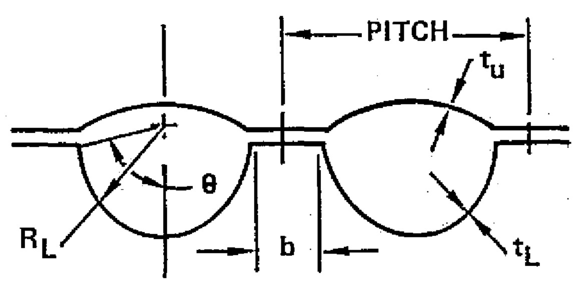
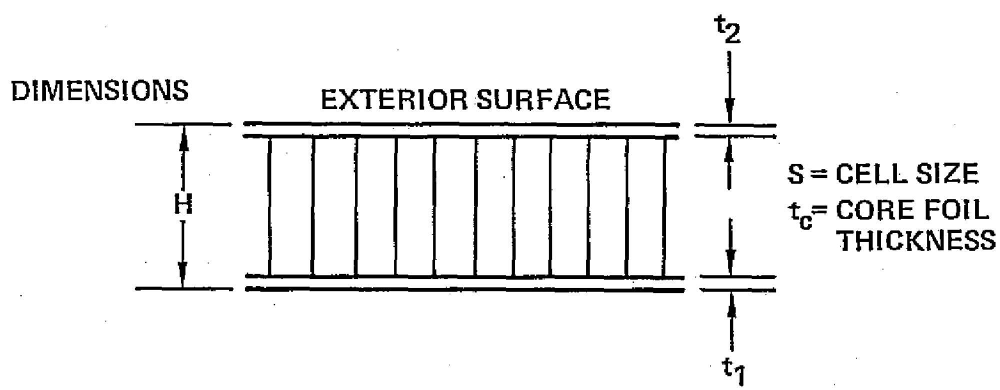


TABLE 12-82. PANEL GEOMETRY AND WEIGHT FOR THE MONOCOQUE PANELS - TASK IIB

DESIGN DATA	POINT DESIGN REGIONS					
	41036		41316		41348	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
RIB	60.0	60.0	40.0	40.0	40.0	40.0
SPAR	21.2	21.2	40.0	40.0	30.0	30.0
DIMENSIONS						
H, in.	.642	.202	1.243	.485	.967	.216
t_1, in.	.026	.023	.056	.062	.037	.042
t_2, in.	.018	.028	.053	.068	.027	.039
t_c, in.	.002	.002	.002	.002	.002	.002
S, in.	.275	.500	.298	.500	.326	.500
WEIGHT DATA						
$\bar{t}$, in.	.052	.052	.124	.133	.075	.082
W, lb/sq. ft.	1.20	1.20	2.850	3.070	1.73	1.89
CRITICAL DESIGN COND.	12	12	12	12	12	12



Conversely, a maximum-weight of 3.07 lb/sq.ft was noted for the lower surface panel at Region 41316. In addition to the panel geometry and weight data, the critical design condition is also specified for each of the point design regions.

Substructure Analysis - The results of the Task I substructure analysis conducted on the chordwise and monocoque arrangements were incorporated into the element definition of the strength design structural model. This data represented typical substructure for each arrangement and included spar caps, spar webs, rib caps, and rib webs. For the chordwise substructure, the spar caps and truss webs are primarily the only load dependent components with the bulkhead webs predominately designed by fuel pressure and the rib caps are minimum design caps. For the monocoque substructure, almost all components were based on minimum design geometry with the exception of the inboard region of the wing tip were the high load intensities dictated greater web thicknesses and cap areas.

No discrete point design weights were defined for the strength designed substructure, but a relatively comprehensive stress analysis was conducted at the model element level using the internal stresses from the strength design run with conservative gross area allowables.

Strength/Stiffness Design

The results of the vibration and flutter anlysis of the strength sized airplane indicated a deficiency in flutter speed for the Mach 0.9 flight condition. This condition is displayed in Figure 12-69 where a flutter speed of 310 KEAS was obtained as compared to the 1.2 V_D criteria of 468 KEAS. Because of this deficiency, a flutter optimization analysis (described in the analytical method section of section 10) was conducted which was focused on incrementing the stiffness of the most efficient wing tip regions. Figure 12-70 presents the five wing tip regions used for the flutter optimization analysis overlayed on the wing tip of the structural model.

The wing tip panel thicknesses used in the definition of the strength design finite element model are shown in Figure 12-71 and represent the thicknesses for both upper and lower surface panels, and the front and rear beam webs. The corresponding wing tip thicknesses resulting from the flutter optimization process (flutter speed equivalent to 1.2 V_D) are shown in Figure 12-72. From a comparison of these two

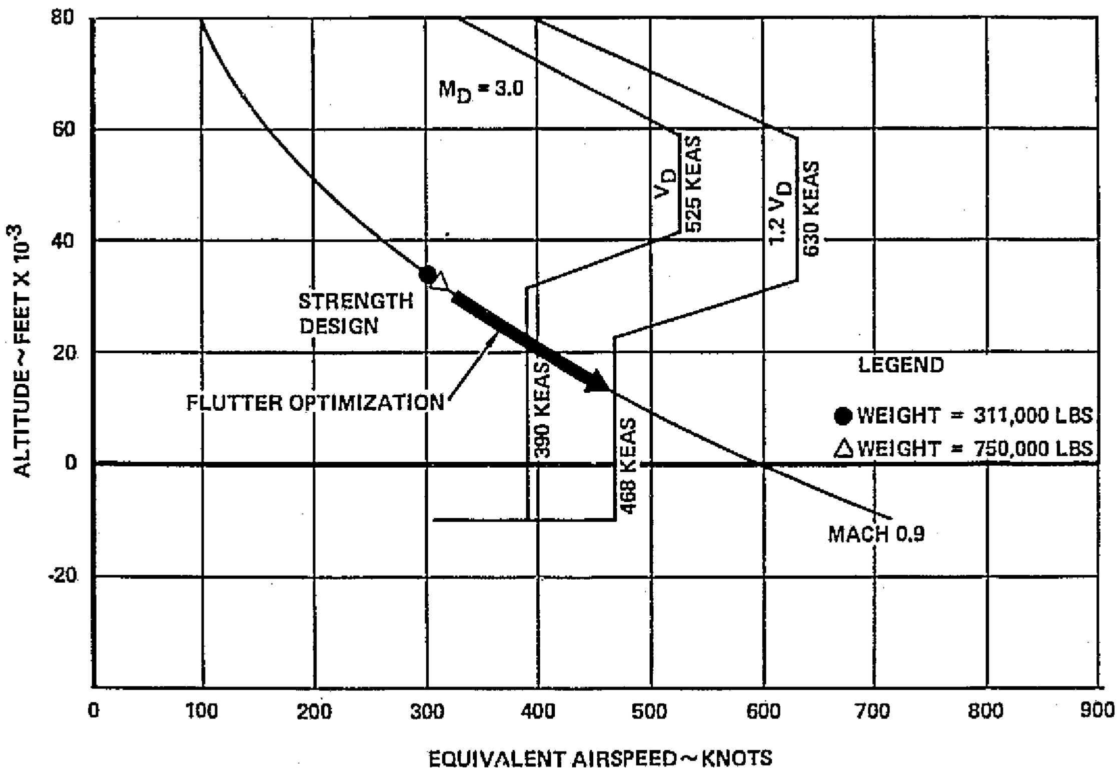


Figure 12-69. Flutter Speeds for Symmetric Bending and Torsion Mode - Task II Strength Design

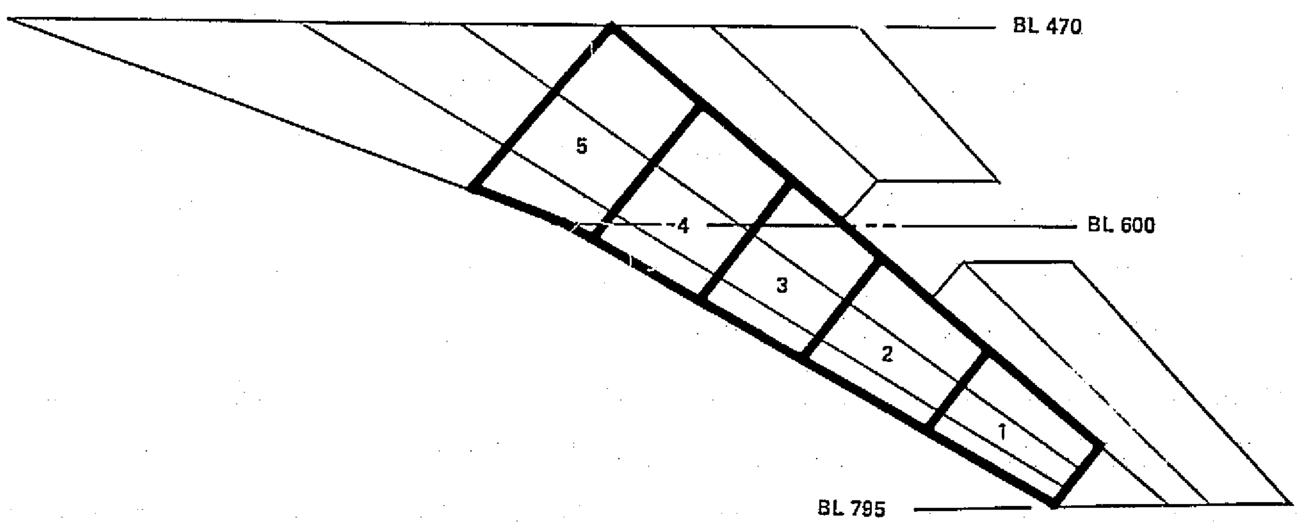
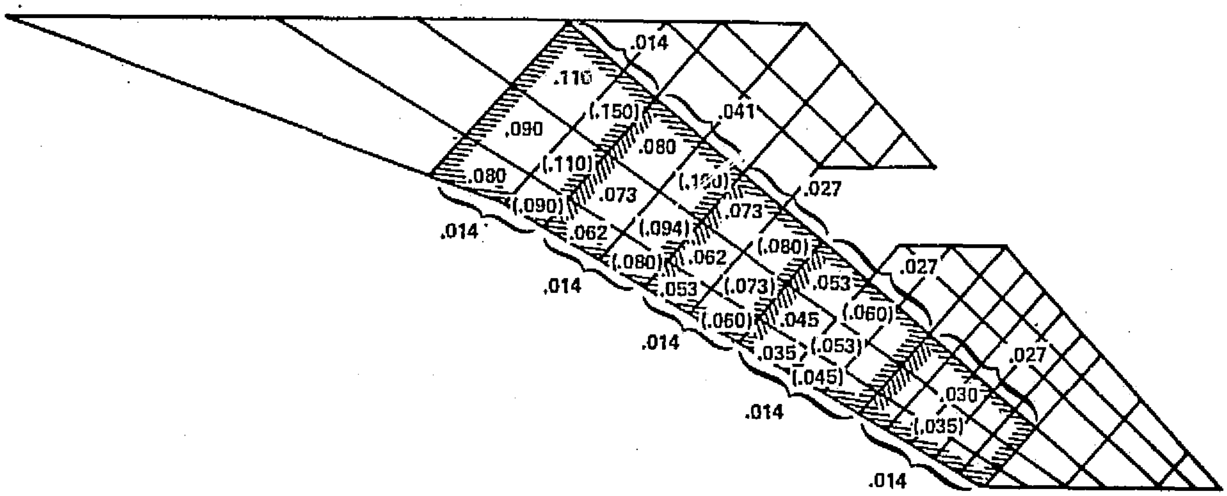


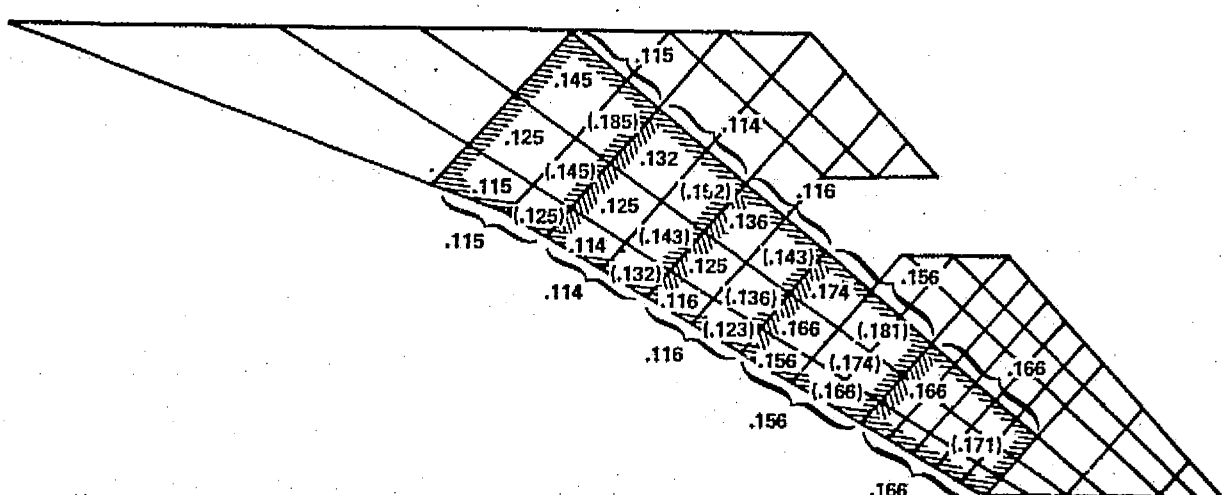
Figure 12-70. Flutter Optimization Design Regions



NOTE:

XXX = UPPER SURFACE EFFECTIVE THICKNESS (IN)
(XXX) = LOWER SURFACE EFFECTIVE THICKNESS (IN)

Figure 12-71. Surface Panel Thickness - Strength Design



NOTE:

XXX = UPPER SURFACE EFFECTIVE THICKNESS (IN)
(XXX) = LOWER SURFACE EFFECTIVE THICKNESS (IN)

Figure 12-72. Surface Panel Thickness - Stiffness Requirements

figures, it can be seen that the outermost regions (Regions 1, 2, and 3 of Figure 12-70) require the greatest change in thickness with Region 1 requiring the maximum change, i.e., an approximate 400-percent increase in surface panel thickness.

Since these flutter optimization results, added wing tip stiffnesses, were generated by incrementing the stiffness matrix from the strength design model, an update in the model section properties was required to obtain a new base stiffness matrix and verify the finding of the flutter optimization process. Thus, the element properties of the 3-D finite element model were revised to reflect the stiffnesses dictated by the flutter analysis with a more favorable material distribution from a design and fabrication standpoint. This task was accomplished by using the basic geometric data (number of elements and their corresponding coordinates) contained in the 3-D structural model, adjusting the section properties of these elements, and calculating the cross-sectional properties (area, center of gravity, and moments of inertia). This process was repeated until the cross sectional properties were equivalent to those required by the flutter optimization analysis. The wing tip panel thicknesses that correspond to the equivalent cross sections are shown in Figure 12-73 and reflect a more favorable material distribution from a design and fabrication standpoint.

A comparison of the panel thicknesses required for the strength, stiffness, and final designs are shown in Figure 12-74 for a wing tip cross section at Region 2 (see Figure 12-70 for location). With reference to this figure, the final design panel thicknesses reflect a uniform distribution of material with the most forward panel on the upper surface indicating a 400-percent increase over the strength design. Similarly, a minimum increase of approximately 200-percent is noted for the lower surface most aftward located panel. The stiffness design reflects a non-uniform material distribution consistent with the methods employed in the flutter optimization process, i.e., incrementing the stiffness matrix of the strength design airplane.

A comparison of the wing tip cross sectional properties for the three designs are shown in Figure 12-75 through 12-78. Figure 12-75 contains a comparison of the centroidal distances along the X-axis, and Figures 12-76 and 12-77 display a comparison of the bending stiffnesses (EI) about the X- and Z-axes, respectively. The final figure (Figure 12-78) presents a comparison of the wing tip torsional stiffness (GJ) for the three designs.

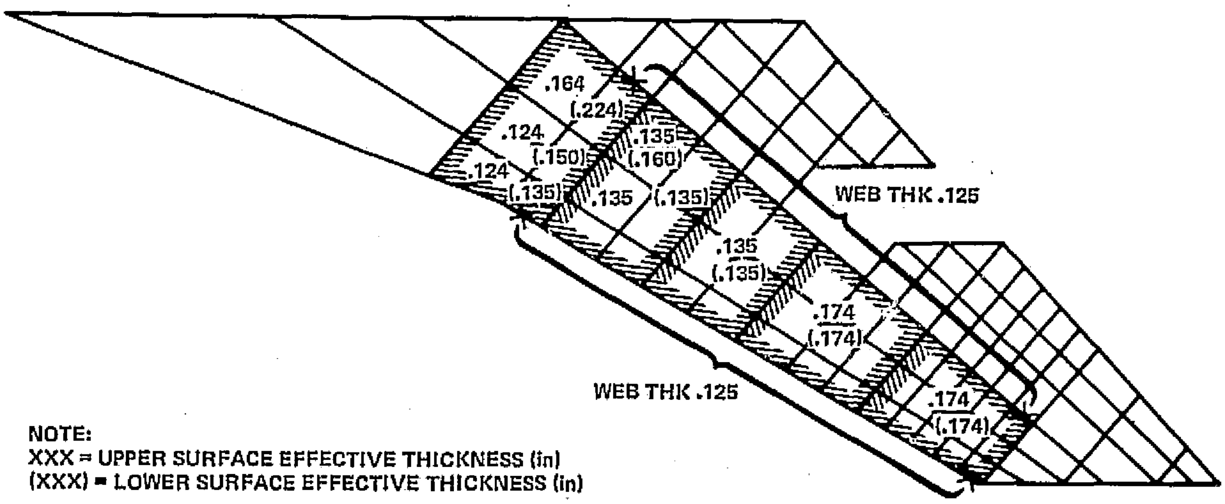


Figure 12-73. Surface Panel Thickness - Final Design

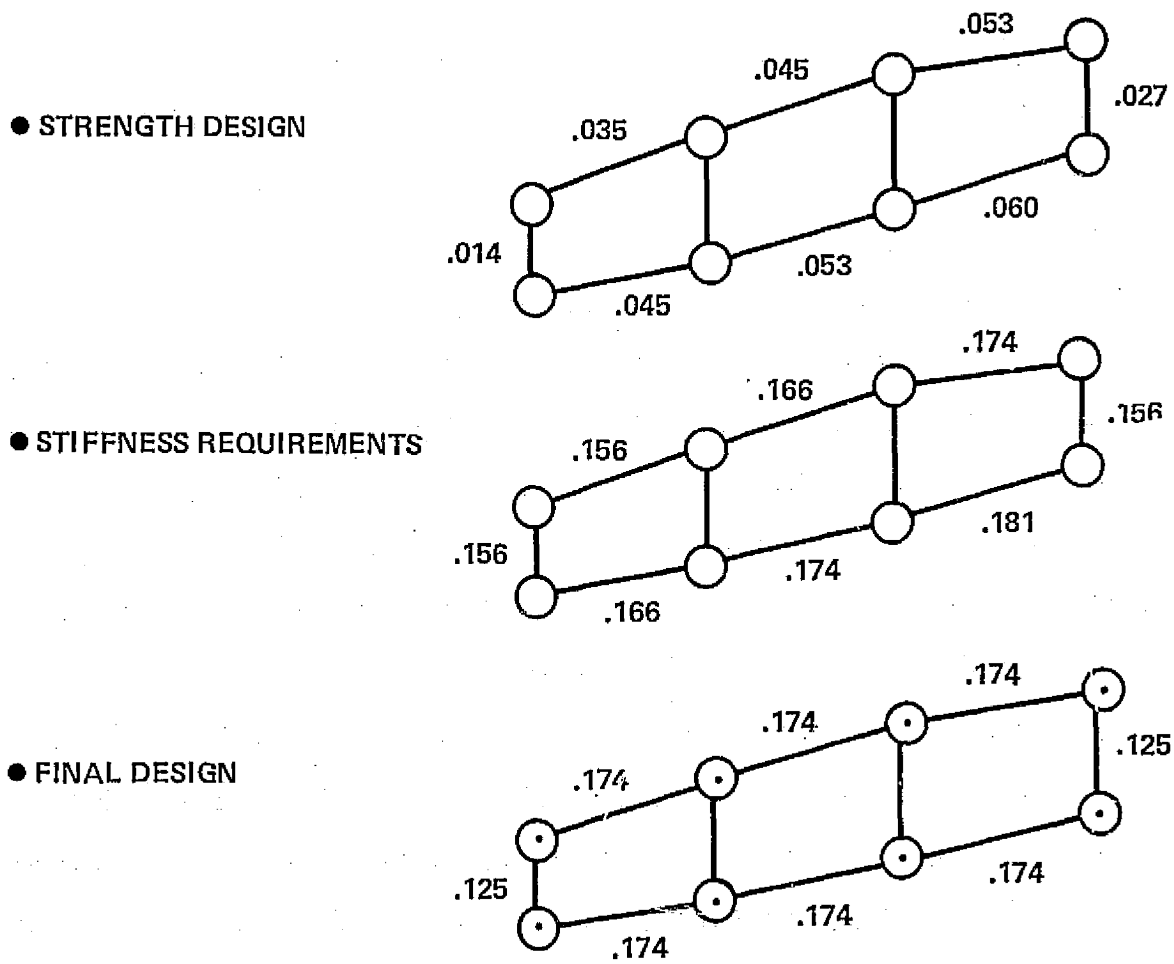


Figure 12-74. Comparison of Cross Sectional Thickness for Wing Tip Region 2

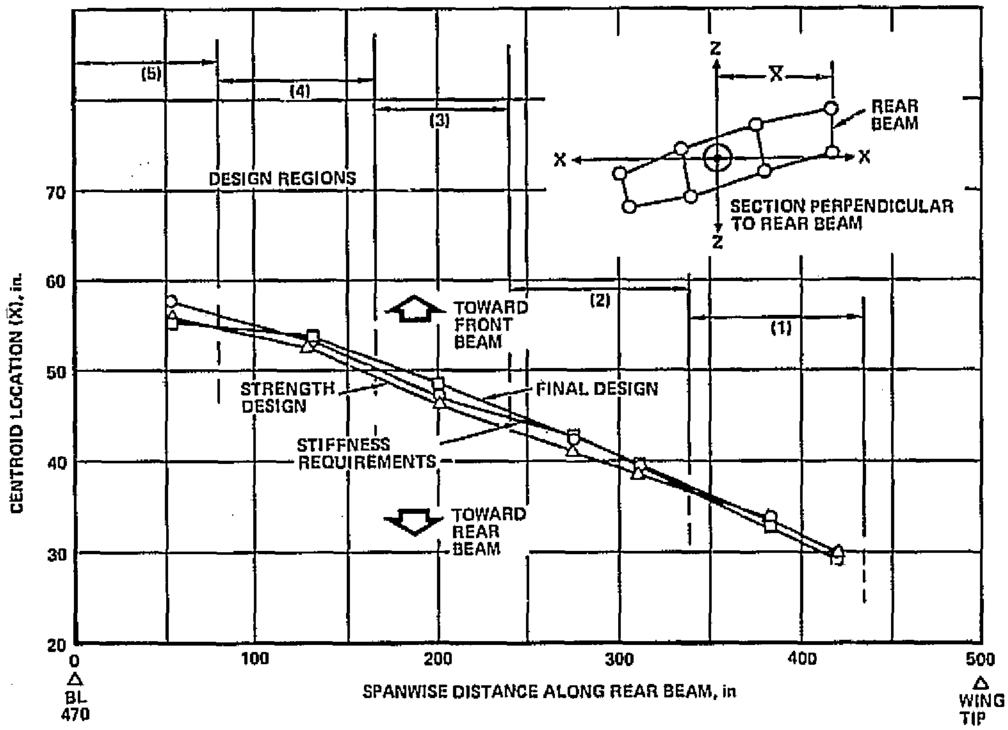


Figure 12-75. Comparison of Wing Tip Center of Gravity

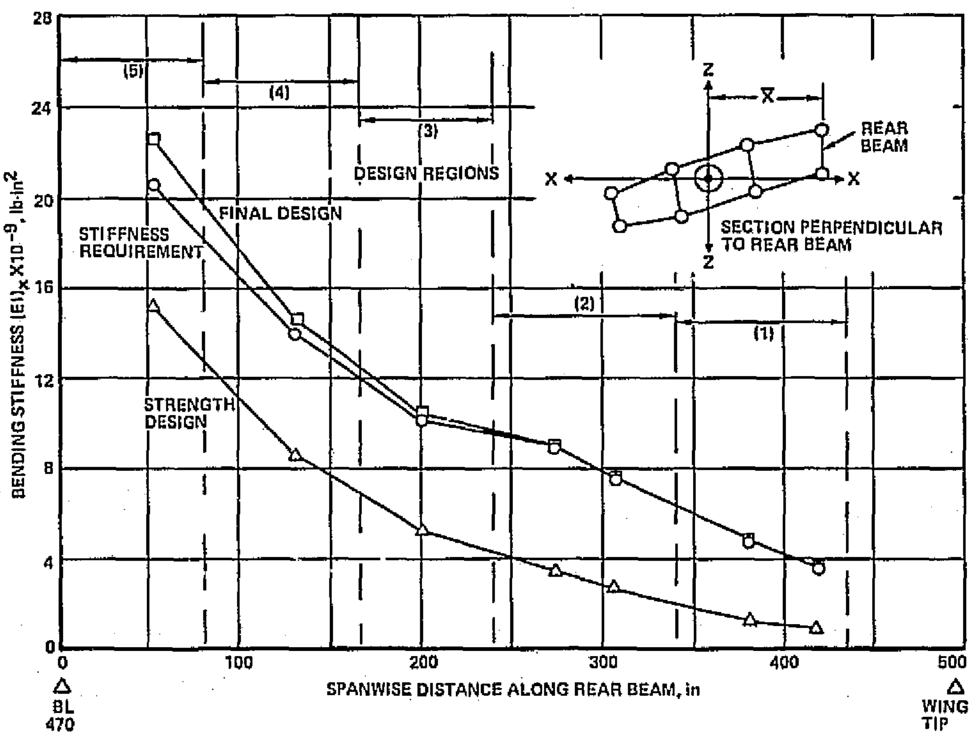


Figure 12-76. Comparison of Wing Tip Bending Stiffness $(EI)_X$

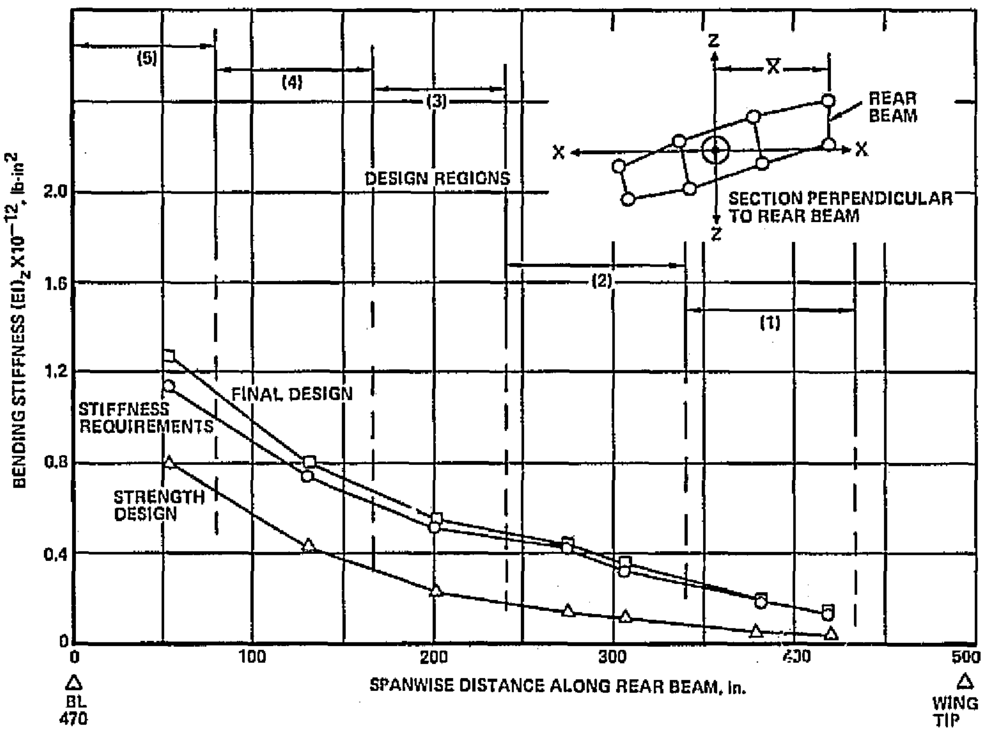


Figure 12-77. Comparison of Wing Tip Bending Stiffness $(EI)_z$

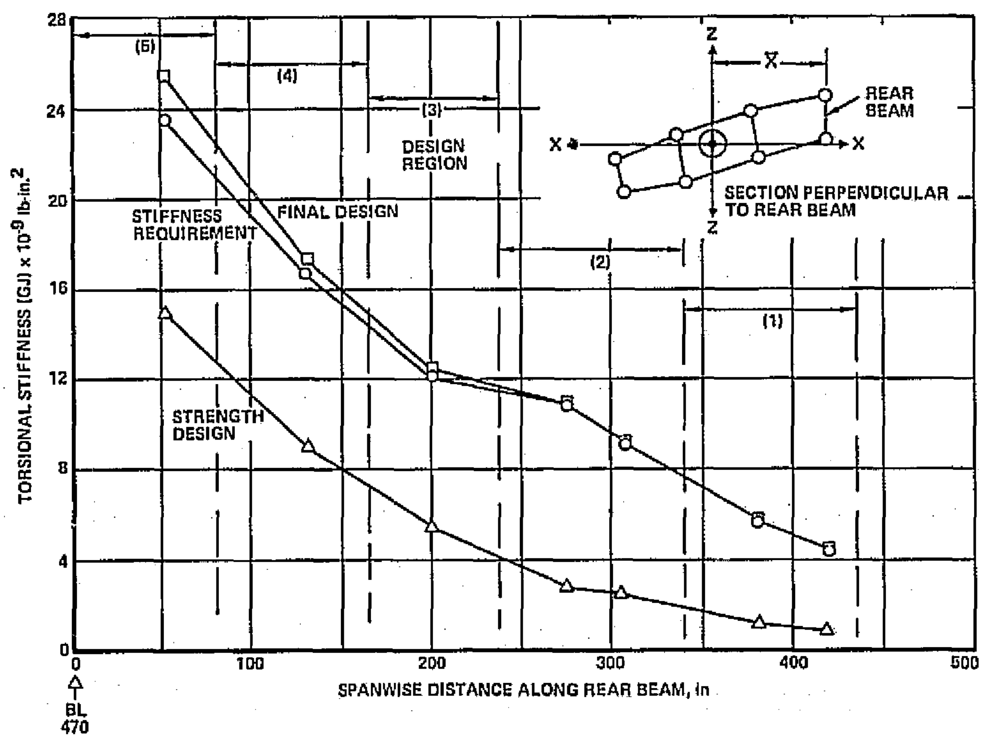


Figure 12-78. Comparison of Wing Tip Torsional Stiffness (GJ)

Point Design Environment - Since the basis for the definition of the loads/temperature environment is the internal loads resulting from the NASTRAN static solution, the section properties for the 3-D structural model were revised to reflect the final design stiffnesses and rerun to obtain the corresponding internal loads. A comparison of the wing surfaces load intensities for the strength and final design airplanes are shown in Table 12-83 for the six wing point design regions. These results are for a symmetrical flight condition at Mach 1.25 and correspond to the static aeroelastic loads analysis conducted using each of the structural models. In addition, the reader should review this table with the specific structural arrangement incorporated in the Task II baseline airplane in mind, i.e., the chordwise stiffened panel concept employed at point design regions 40322, 40236, and 40536, and the honeycomb sandwich concept at regions 41036, 41316, and 41348.

In general, the chordwise panel inplane load intensities (Nx and Nxy) are greater for the strength design than those of the final design; whereas, the final design airplane experiences the higher spar cap loads (Ny). For the regions which incorporate the honeycomb sandwich surface panels, the combined loads (Nx, Ny, and Nxy) are generally higher for the final design with the exception of slightly lower shear values experienced by the inboard panels (41036 and 41316).

The point design environments (airloads, thermal strains, pressures, and temperatures) for the final design airplane were identical to those reported for the strength design except for the airloads which reflected the additional NASTRAN internal load run. An example of this environment is presented in Table 12-84 for the symmetrical flight conditions at Mach 0.90 and Mach 1.25. A detail description of the point design environments for the final design airplane is contained in Section 11 of this report.

Panel Analysis - The wing panel geometry calculated for the strength sized airplane were reviewed with respect to the results of the flutter optimization study and the ensuing internal loads run. The results of this review indicated the chordwise panel concepts (circular-arc convex-beaded concept), which are predominately designed by normal pressure, experienced a relatively slight decrease in inplane loads due to the change in airframe stiffness; thus, it was felt no additional analysis was warranted and the strength-size panel geometry was incorporated into the final design airplane.

TABLE 12-83. COMPARISON OF WING SURFACE LOAD INTENSITIES - TASK IIB, MACH 1.25 LOAD CONDITION

PANEL IDENTIFICATION		*LOAD INTENSITY (ULTIMATE), LBS/IN.				
			HYBRID (STRENGTH)		HYBRID (FINAL)	
REGION	NUMBER	DIRECTION	UPPER	LOWER	UPPER	LOWER
WING-FORWARD	40322	Nx	-151	597	-242	434
		Ny	-1106	1400	-1032	1425
		Nxy	130	215	102	166
WING-AFT BOX	40236	Nx	-67	246	-183	62
		Ny	-14650	15196	-16456	16622
		Nxy	453	367	491	781
	40536	Nx	-1073	1099	-831	699
		Ny	-14303	14014	-16372	15508
		Nxy	1495	1599	1615	1646
	41036	Nx	-1812	1297	-2464	1898
		Ny	-4220	3588	-5645	4697
		Nxy	2106	1909	1915	1812
WING TIP	41316	Nx	-1638	1405	-1931	1656
		Ny	-12407	11188	-13240	11333
		Nxy	4009	2990	4072	2739
	41348	Nx	-1207	1379	-1200	1431
		Ny	-6897	6657	-9006	8090
		Nxy	2284	2281	2666	2556

*LOAD CONDITIONS:

TASK II-B CONDITION 12: MACH 1.25, nz = 2.5, W = 690,000 LB, Ve = 294 KEAS

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TABLE 12-84. WING POINT DESIGN ENVIRONMENT, FINAL DESIGN - TASK IIB, MACH 0.90 (V_C) AND 1.25 (V_S) LOAD CONDITIONS

CONDITION (8) SYMMETRICAL FLIGHT, STEADY MANEUVER AT MACH 0.90 (V_C), n_z = 2.5

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION 40322 UPPER SURFACE	40322 LOWER SURFACE	41316 UPPER SURFACE	41316 LOWER SURFACE	41348 UPPER SURFACE	41348 LOWER SURFACE	40236 UPPER SURFACE	40236 LOWER SURFACE	40536 UPPER SURFACE	40536 LOWER SURFACE	41036 UPPER SURFACE	41036 LOWER SURFACE
AIR LOADS	Nx	LB/IN	- 219	+ 468	- 1,478	+1,237	- 856	-1,028	+ 15	+ 451	- 315	+ 279	-1,562	+1,083
	Ny	LB/IN	-1,049	+1,381	-10,106	+8,620	-6,598	+5,862	-14,311	-14,762	-14,410	+14,085	-4,725	+4,032
	Nxy	LB/IN	75	166	3,730	2,842	2,608	2,641	272	370	1,159	1,300	1,773	1,454
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	PSI	- 1.40	- 0.30	- 6.75	- 3.76	- 3.68	- 0.30	- 0.99	- 0.44	- 1.35	- 0.49	- 2.10	- 0.36
	FUEL	PSI	- 4.91	- 8.70	0	0	0	0	- 5.01	- 9.24	- 4.50	- 4.50	0	0
	NET	PSI	- 6.31	- 9.00	- 6.75	- 3.76	- 3.68	- 0.30	- 6.00	- 9.68	- 5.85	- 4.99	- 2.10	- 0.36
TEMPERATURE	T_{AV}	°F	47	52	48	41	44	33	47	53	45	45	48	41
	ΔT	°F	+ 37	+ 41	+ 27	+ 15	- 20	- 8	+ 37	- 40	+ 30	- 36	+ 27	+ 15

CONDITION (12) SYMMETRICAL FLIGHT, STEADY MANEUVER AT MACH 1.25 (V_S), n_z = 2.5

ULTIMATE DESIGN LOADS	ITEM	UNITS	POINT DESIGN REGION 40322 UPPER SURFACE	40322 LOWER SURFACE	41316 UPPER SURFACE	41316 LOWER SURFACE	41348 UPPER SURFACE	41348 LOWER SURFACE	40236 UPPER SURFACE	40236 LOWER SURFACE	40536 UPPER SURFACE	40536 LOWER SURFACE	41036 UPPER SURFACE	41036 LOWER SURFACE
AIR LOADS	Nx	LB/IN	- 242	+ 434	- 1,931	- 1,656	-1,200	+1,431	- 183	+ 62	- 831	+ 699	-2,464	+1,898
	Ny	LB/IN	-1,032	+1,425	-13,240	+11,333	-9,006	+8,390	-16,456	+16,622	-16,372	+15,508	-5,645	+4,697
	Nxy	LB/IN	102	166	4,072	2,739	2,666	2,556	491	781	1,615	1,646	2,334	1,812
THERMAL STRAIN	ϵ_x	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_y	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
	ϵ_{xy}	IN/IN	0	0	0	0	0	0	0	0	0	0	0	0
PRESSURE	AERO	PSI	- 1.47	+ 0.06	- 4.98	- 0.26	- 5.07	+ 0.96	- 3.03	- 1.20	- 1.27	- 0.26	- 1.27	+ 0.11
	FUEL	PSI	- 4.97	- 8.89	0	0	0	0	- 5.08	- 9.31	- 4.50	- 4.50	0	0
	NET	PSI	- 6.44	- 8.83	- 4.98	- 0.26	- 5.07	- .96	- 8.11	- 10.51	- 5.77	- 4.76	- 1.27	+ 0.11
TEMPERATURE	T_{AV}	°F	RT	RT	RT	RT	RT	RT	RT	RT	RT	RT	RT	RT
	ΔT	°F	0	0	0	0	0	0	0	0	0	0	0	0

NOTES: (1) A 1.25 FACTOR HAS BEEN APPLIED TO THE THERMAL STRAIN WHEN THE SIGN IS SAME AS THE AIRLOAD SIGN, OTHERWISE NO FACTOR APPLIED.

(2) PRESSURE SIGN CONVENTION: NEGATIVE = SUCTION

The point design regions utilizing the honeycomb sandwich concept (regions 41036, 41316, and 41348) experienced relatively large changes in inplane loads due to the modification of the airframe stiffness. For regions 41316 and 41348 the increased panel thickness required to meet the flutter speed was much greater than the corresponding change in internal loads due to the aeroelastic effect, e.g., for the upper surface at region 41348, the thickness increased 112-percent while the inplane loads, as characterized by the spanwise load (Ny), increased only 30-percent. Hence the surface panels at regions 41316 and 41348 are stiffness designed and their panel geometry and weight data are presented in Table 12-85. For the honeycomb sandwich panel at region 41036, located inboard of the wing tip, the fail-safe criteria required major changes in the panel proportions, i.e., 50- and 61-percent changes in face sheet thickness over the strength design for the upper and lower panels, respectively. Section 13 presents the analysis and required panel geometry for this region.

<u>Substructure Analysis</u> - Typical substructure was investigated for application to the structural arrangement of the final design airplanes. This substructure included: rib caps, rib webs, spar caps, and spar webs. In addition, nonoptimum factors applicable to each concept were added. The strength design geometry and weights were used for those substructure components experiencing slight or no change in applied load.

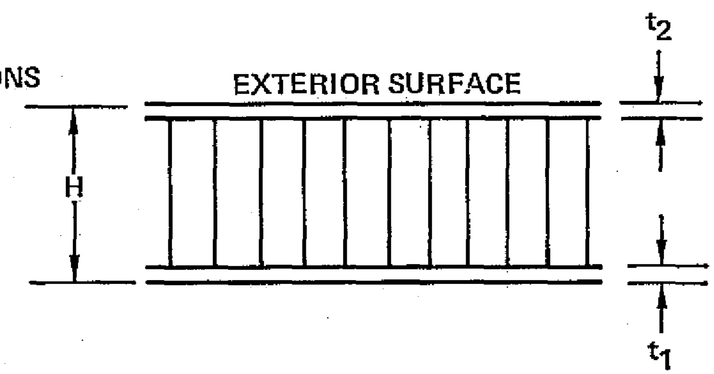
The major weight components for the substructure are the spar caps and these results are presented in this section. As with the strength design, the chordwise point design regions (40322, 40236, and 40536) incorporate submerged metallic spar caps with B/PI reinforcement; whereas, the monocoque regions (41036, 41316, and 41348) use an all metal design incorporating a densified core insert with mechanical fasteners for attachment to the surface panels.

For the composite reinforced spar caps, the maximum spanwise tension and compression loads intensities are shown in Table 12-86. In addition, the total load on the caps (Ny times b) are defined at each point design region. The stress analysis was conducted using the same methods and allowables as defined for the Task I composite substructure analysis and is shown in Table 12-87. Figure 12-59 presented the allowable axial stress (tension or compression) for the B/PI reinforced caps. As a result of the stress analysis, the spar cap geometry and corresponding weight are shown in Table 12-88. The weight of the composite reinforced spar caps ranged from

TABLE 12-85. PANEL GEOMETRY AND WEIGHT DATA FOR THE FINAL DESIGN MONOCOQUE PANELS - TASK IIB

	POINT DESIGN REGIONS			
	41316		41348	
	UPPER	LOWER	UPPER	LOWER
SPACING, in.				
RIB	40.0	40.0	40.0	40.0
SPAR	40.0	40.0	30.0	30.0
DIMENSIONS				
H, in.	1.00	.500	1.00	.500
t_1, in.	.062	.075	.068	.068
t_2, in.	.062	.075	.068	.068
t_c, in.	.002	.002	.002	.002
S, in.	.500	.500	.500	.500
WEIGHT DATA				
$\bar{t}$, in.	.131	.153	.143	.139
W, lb./sq. ft.	3.02	3.52	3.29	3.20
CRITICAL DESIGN COND.	FLUTTER	FLUTTER	FLUTTER	FLUTTER

DIMENSIONS



S = CELL SIZE

t_c = CORE FOIL THICKNESS

TABLE 12-86. SPAR CAP APPLIED LOADS FOR THE BASIC WING REGIONS, TASK IIB

POINT DESIGN REGION	WING SURFACE	SPAR SPACING b, (in.)	MAXIMUM SPANWISE(1) TENSION LOAD			MAXIMUM SPANWISE(1) COMPRESSION LOAD			DESIGN LOADS (ULT.)		
			COND.	N_y (lb/in.)	P_y (kips)2	COND.	N_y (lb/in.)	P_y (kips)(2)	FACTOR(3)	P_T (kips)(4)	P_C (kips)(4)
40322	UPPER	22.7	14	400	9.1	15	-1,100	-25.0	1.1	10.0	-27.5
	LOWER	22.7	12	1,400	31.8	14	-600	-13.6	1.1	35.0	-15.0
40236	UPPER	21.2	14	8,100	171.4	12	-16,400	-348.9	1.1	190.2	-387.3
	LOWER	21.2	12	16,600	352.4	14	-7,800	-164.4	1.1	391.2	-182.5
40536	UPPER	21.2	14	7,500	158.8	12	-16,400	-347.1	1.1	176.3	-385.3
	LOWER	21.2	12	15,500	328.8	14	-6,600	-140.2	1.1	364.9	-155.6
41036	UPPER	21.2	14	2,500	53.7	12	-5,600	-119.7	1.2	62.3	-138.8
	LOWER	21.2	12	4,700	99.6	14	-2,000	-42.9	1.2	115.5	-49.8

1. CONDITION DESCRIPTION AND LOAD INTENSITIES PER SECTION 11
2. CAP LOAD (P_Y) = $N_y \times b$
3. CORRECTION FACTOR TO ACCOUNT FOR SUBMERGED CAPS
4. P_T = MAXIMUM TENSILE LOAD

   P_C = MAXIMUM COMPRESSIVE LOAD

TABLE 12-87. SPAR CAP STRESS ANALYSIS FOR THE BASIC WING REGIONS, TASK IIB

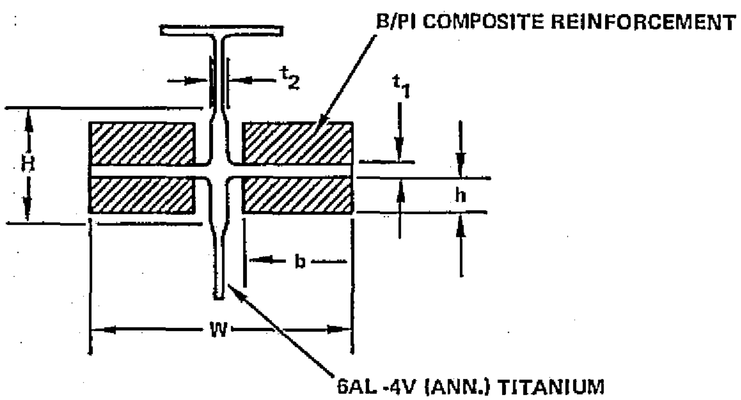
POINT DESIGN REGION	SURFACE	SPAR SPACING (in.)	DESIGN LOAD		AREA			$f_y^{C,T}$ (ksi)	% COMPOSITE (A_C/A_T)	$F_y^{C,T}$ (ksi)	MARGIN OF SAFETY
			COND. NO.	P_{ULT} (kips)	A_M (in.^{2})	A_C (in.^{2})	A_T (in.^{2})				
40322	UPPER	22.7	15	-27.5	0.24	—	0.24	-114.6	—	-131.0	0.14
	LOWER	22.7	12	35.0	0.40	—	0.40	87.5	—	90.0	0.03
40236	UPPER	21.2	12	-387.3	0.45	1.51	1.96	-197.6	77	-198.0	0.00
	LOWER	21.2	12	391.2	0.45	2.50	2.95	132.6	85	139.0	0.05
40536	UPPER	21.2	12	-385.3	0.45	1.50	1.95	-197.6	77	-198.0	0.00
	LOWER	21.2	12	364.9	0.45	2.30	2.75	13.7	84	138.0	0.04
41036	UPPER	21.2	12	-138.8	0.45	0.41	0.86	-161.4	48	-162.0	0.00
	LOWER	21.2	12	115.5	0.45	0.50	0.95	121.6	53	122.0	0.00

NOTES:

1. $f_y^{C,T} = P_{ULT} \div A_T$
2. ALLOWABLE STRESSES ($F_y^{C,T}$) PER FIGURE 12-59
3. MARGIN OF SAFETY = $(F_y^{C,T} \div f_y^{C,T}) - 1.0$

TABLE 12-88. SPAR CAP GEOMETRY FOR THE BASIC WING REGIONS, TASK IIB

POINT DESIGN REGION	SPAR SPACING (in.)	SPAR CAP DIMENSIONS h (in.)	b (in.)	H (in.)	W (in.)	t_1 (in.)	t_2 (in.)	AREA A_M (in.^{2})	A_C (in.^{2})	UNIT WEIGHT w (lb/sq.ft)
40322										
UPPER	22.7	–	–	–	1.50	.16	–	0.24	–	0.24
LOWER	22.7	–	–	–	1.50	.27	–	0.40	–	0.41
40236										
UPPER	21.2	.38	1.00	1.20	2.50	.12	.12	0.45	1.51	1.23
LOWER	21.2	.62	1.00	1.20	2.50	.12	.12	0.45	2.50	1.71
40536										
UPPER	21.2	.38	1.00	1.20	2.50	.12	.12	0.45	1.50	1.22
LOWER	21.2	.58	1.00	1.20	2.50	.12	.12	0.45	2.30	1.61
41036										
UPPER	21.2	.10	1.00	1.20	2.50	.12	.12	0.45	0.41	0.69
LOWER	21.2	.12	1.00	1.20	2.50	.12	.12	0.45	0.50	0.73



w = EQUIVALENT UNIT PANEL WEIGHT, lb/sq.ft.
$= (\rho_M \times A_M + \rho_c \times A_c) \times 144/\text{SPAR SPACING}$

WHERE:

ρ_M = TITANIUM DENSITY = 0.160 lb/in.^{3}

A_M = TITANIUM AREA = $(H - t_1)t_2 + W \times t_1$

ρ_C = BORON/POLYIMIDE DENSITY = 0.072 lb/in.^{3}

A_C = BORON/POLYIMIDE AREA = $4 \times b \times h$

a minimum of 0.024 lb/sq.ft for the upper cap at region 40322 to a maximum of 1.23 lb/sq.ft for the corresponding cap at region 40236.

The spar cap geometry and weight for the honeycomb sandwich panels at regions 41316 and 41348 are shown in Table 12-89 and a sketch of this design was previously presented in Figure 12-38. This table displays the cross-sectional area of each of the spar cap components (doublers, densified core, and web attachment) as well as the total weight. A minimum weight of 0.12 lb/sq.ft is indicated for the spar caps at region 41316 with a maximum weight of 0.16 lb/sq.ft noted for the upper spar caps at region 41348.

A summary table of wing spar cap results is shown in Table 12-90 for the six wing point design regions. This table summarizes the material system, panel dimensions, cap areas, and unit weights for each regions.

In addition to the primary Boron/polyimide (B/PI) material system used in the design of the chordwise spar caps an alternate Boron/aluminum (B/Al) design was evaluated for back-up purposes. The spars at region 40536 were selected for this investigation due to the relatively high spanwise loading.

The results of the weight/strength analysis conducted on the B/Al spar caps are shown in Table 12-91 and includes the corresponding B/PI design data for comparison purposes. Similar to the B/PI design, a minimum area titanium substrate of 0.45 in.^{2} was considered in the stress analysis. The B/Al allowable stresses (tension and compression), and the modulus and density are presented in Figure 12-79 as a function of the Boron/Aluminum fraction of total area. With reference to the tension allowable, no pertinent fatigue data was found concerning the fatigue life of the combined B/Al and titanium material system. Thus, the tension cut-off stress for the combined system was based on the fatigue allowable of the Boron/Aluminum material obtained from data published in References 8 and 9. The B/Al tension cutoff stress used for this investigation corresponding to a stress ratio (R) of 0.4.

In conclusion, the B/Al design for the upper spar caps show a 7-percent weight savings over the B/PI design. Conversely, the B/Al material is the heaviest design for the lower surface caps indicating a weight penalty of 30-percent. Combining this data results in an overall weight penalty of approximately 14-percent for the alternate B/Al spar cap design.

TABLE 12-89. SPAR CAP GEOMETRY FOR THE WING TIP REGIONS, TACK IIB

POINT DESIGN REGION	WING SURFACE	SPAR SPACING b (in.)	DOUBLER			WEB ATTACH			CORE					$\bar{t}$ (in.^{2}/in.	W (lb/sq.ft.)
			t_{do} (in.)	t_{di} (in.)	A_D (in.^{2})	A_B (in.^{2})	A_I (in.^{2})	A_W (in.^{2})	$\bar{h}$ (in.)	t_c (in.)	S_D (in.)	S_P (in.)	ΔA		
41316	UPPER	40.0	.023	.020	.086	.08	.08	.08	.853	.002	.12	.50	.043	.005	.115
	LOWER	40.0	.024	.020	.088	.08	.08	.08	.326	.002	.12	.50	.016	.005	.115
41348	UPPER	30.0	.023	.020	.086	.08	.08	.08	.841	.002	.12	.50	.043	.007	.161
	LOWER	30.0	.023	.020	.086	.08	.08	.08	.341	.002	.12	.50	.017	.006	.138

- DOUBLER AREA (A_D)

$A_D = 2.00\,(t_{do} + t_{di})$

t_{do} = OUTER SKIN DOUBLER

t_{di} = INNER SKIN DOUBLER

- WEB ATTACHMENT AREA (A_W)

$A_W = (A_B + A_I) \times 1/2 = .10$ in.^{2}

A_B = BULKHEAD ATTACHMENT = .08 in.^{2}

A_I = INTERMEDIATE SPAR ATTACH = .12 in.^{2}

- DENSIFIED CORE INSERT

$\Delta A = 4.0\,\bar{h}\,\Delta(t_c/S)$

$\bar{h}$ = CORE HEIGHT

t_c = CORE FOIL THICK

S = CELL SIZE

$(t_c/S$ = INSERT (t_c/S) - PANEL (t_c/S)

- EQUIVALENT PANEL VALUES

$\bar{t} = 1/b(A_D + A_W + \Delta A)$

$W = 23.04 \times \bar{t}$

TABLE 12-90. SUMMARY OF WING SPAR CAP RESULTS, TASK IIB

POINT DESIGN REGION	CAP LOCATION	CAP DESIGN	SPAR SPACING b (in.)	CAP AREA (in.^{2})			CAP WEIGHT, W (lb./sq.ft.)
				A_C	A_M	A_{TOTAL}	
40322	UPPER	ALL METAL	22.7	–	0.24	0.24	0.24
	LOWER	6Al-4V Ti CAP	22.7	–	0.40	0.40	0.41
40236	UPPER	6Al-4V Ti CAP WITH B/PI REINF	21.2	1.51	0.45	1.96	1.23
	LOWER		21.2	2.50	0.45	2.95	1.71
40536	UPPER	6Al-4V Ti CAP WITH B/PI REINF.	21.2	1.50	0.45	1.95	1.22
	LOWER		21.2	2.30	0.45	2.75	1.61
41036	UPPER	6Al-4V Ti CAP WITH B&PI REINFORCEMENT	21.2	0.41	0.45	0.86	0.69
	LOWER		21.2	0.50	0.45	0.95	0.73
41316	UPPER	ALL METAL 6Al-4V Ti CAP	40.0	–	0.21	0.21	0.12
	LOWER		40.0	–	0.18	0.18	0.12
41348	UPPER	ALL METAL 6Al-4V Ti CAP	30.0	–	0.21	0.21	0.16
	LOWER		30.0	–	0.18	0.18	0.14

NOTES:

A_C = COMPOSITE AREA

A_M = METAL AREA

$A_{TOTAL} = A_C + A_M$

W = EQUIVALENT SURFACE PANEL WEIGHT, lb/sq.ft.; $144(A_c\rho_C + A_M\rho_M)/b$

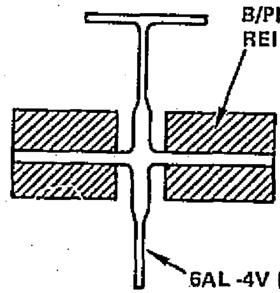
ρ_C = COMPOSITE (B/PI) DENSITY; .072 lb/in.^{3}

ρ_M = METAL (6Al-4V) DENSITY; .160 lb/in.^{3}

b = SPAR SPACING

TABLE 12-91. WEIGHT COMPARISON OF THE ALTERNATE COMPOSITE REINFORCED SPAR CAP DESIGN, TASK IIB

POINT DESIGN REGION	SURFACE	SPAR SPACING (in.)	SPAR DESIGN	DESIGN LOADS COND. NO.	DESIGN LOADS P_{ULT} (kips)	AREA A_M (in.^{2})	AREA A_C (in.^{2})	AREA A_T (in.^{2})	$f_y^{C,T}$ (ksi)	% COMPOSITE	$F_y^{C,T}$ (ksi)	MARGIN OF SAFETY	UNIT WEIGHT w (lb./sq.ft)
40536	UPPER	21.2	B/PI	12	-385.3	0.45	1.50	1.95	-197.6	77	-198.0	0.00	1.22
	LOWER	21.2	REINF.	12	364.9	0.45	2.30	2.75	132.7	84	138.0	0.04	1.61
	UPPER	21.2	B/AI	12	-385.3	0.45	0.95	1.40	-275.0	68	-275.0	0.00	1.13
	LOWER	21.2	REINF.	12	364.9	0.45	2.38	2.83	129.0	84	129.0	0.00	2.09



w = EQUIVALENT SURFACE PANEL UNIT WEIGHT, lb./sq.ft.

$= (\rho_C A_C + \rho_M A_M) \times 144/\text{SPAR SPACING}$

WHERE:

ρ_C = COMPOSITE DENSITY = .099 lb./in.^{3}(B/Al)

= .072 lb./in.^{3}(B/PI)

ρ_M = TITANIUM DENSITY = .160 lb./in.^{3}

A_M, A_C = AREA, in.^{2}

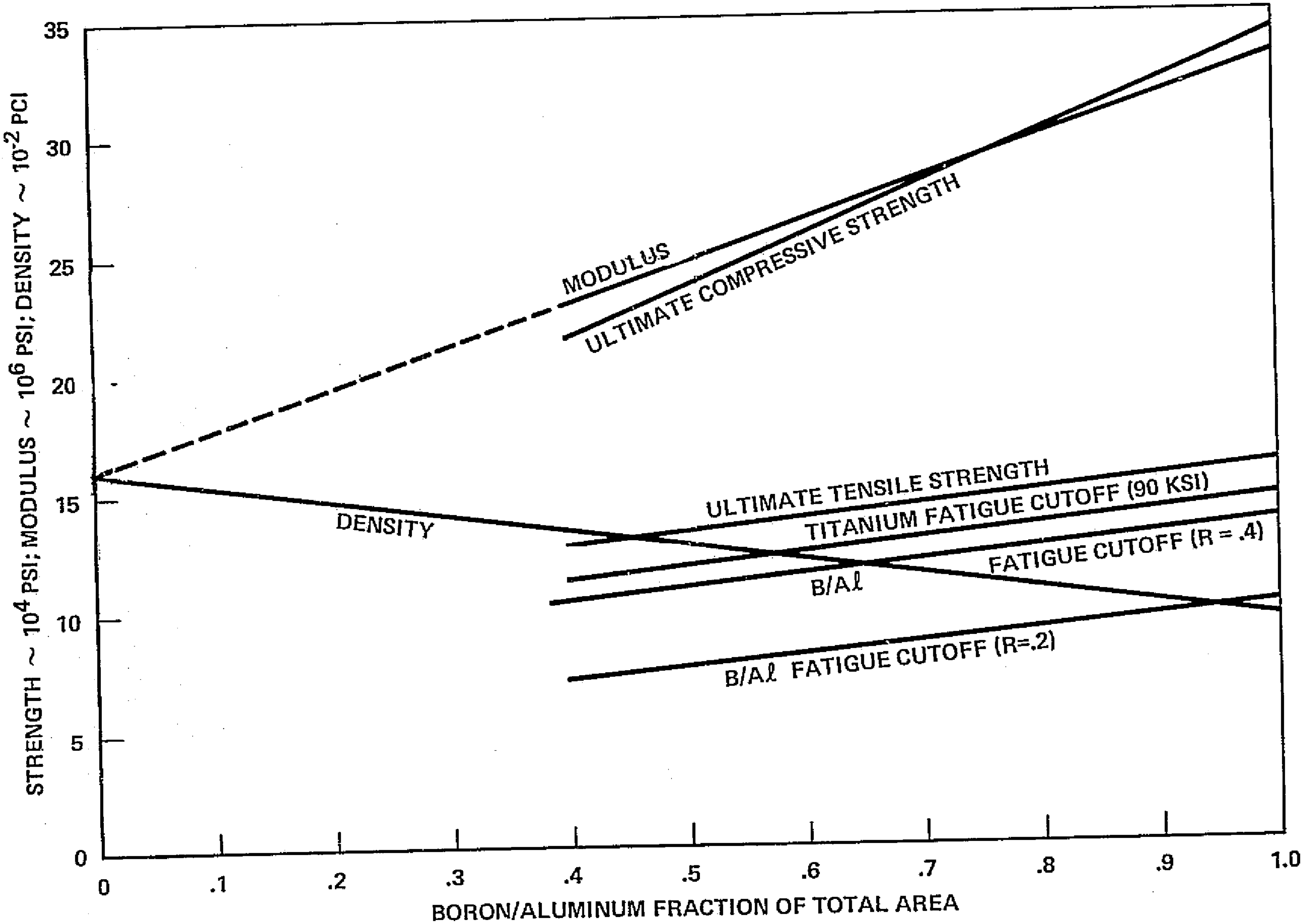


Figure 12-79. Strength, Stiffness and Density of Titanium Reinforced with Boron-Aluminum

Wing Box Unit Weights - A compilation of the component and total wing box unit weights are shown in Table 12-92 for the Task II hybrid structural arrangement. These weights reflect the results of the iterative design cycle conducted on the strength and strength/stiffness designs. The weight penalties associated with the remaining disciplines included in the structural analysis (fail-safe, sonic fatigue, etc.) are reported in their respective sections and are not reflected in the detail weights of Table 12-92.

For the chordwise design regions (40322, 40236, and 40536), a minimum-weight of 3.80 lb/sq.ft occurs at region 40322 and a maximum-weight of 6.99 lb/sq.ft is noted at region 40536. The remaining chordwise region at 40236 has a unit weight of 6.79 lb/sq.ft.

With respect to the regions which incorporate the monocoque design (41036, 41316, and 41348), region 41036 is the minimum-weight region with a unit weight of 1.60 lb/sq.ft followed by regions 41316 and 41348 which have weights of 7.37 lb/sq.ft and 7.44 lb/sq.ft, respectively.

FUSELAGE STRUCTURAL ARRANGEMENT - TASK IIB

The Task II fuselage analysis was conducted using the most promising fuselage concepts surviving the Task I Analytical Design Studies. These concepts included both the zee-stiffened and closed hat-stiffened panel configurations. The zee-stiffened configuration is applicable in the lightly loaded forebody region and the hat-stiffened concept is used for the higher loaded midbody and aftbody regions. Floating zee-shaped frames with skin shear ties were the frame concept considered. These structural concepts are shown in Figure 12-80. In addition, the panel configurations were investigated for both metallic and composite reinforced material systems.

Similar to Task I, the analysis was conducted at four fuselage point design regions using the internal loads resulting from the 3-D structural model redundant analysis. Section 9 contains a detail description of the model, model input data, and the resultant load intensities.

TABLE 12-92. DETAIL WING WEIGHTS FOR THE TASK II HYBRID STRUCTURAL ARRANGEMENT

POINT DESIGN REGION	40322	40236	40536	41036	41316	41348
SPACING (in.)						
SPAR	22.70	21.20	21.20	21.20	40.00	30.00
RIB	60.00	60.00	60.00	60.00	40.00	40.00
PANELS						
UPPER	0.76	0.83	1.34	1.20	3.02	3.29
LOWER	0.95	1.11	1.05	1.20	3.52	3.20
Σ	(1.71)	(1.94)	(2.39)	(2.40)	(6.54)	(6.49)
RIB WEBS						
BULKHEAD	0.30	0.28	0.24	0.13	0.19	0.10
TRUSS	0.07	0.24	0.23	0.11	—	—
Σ	(0.37)	(0.52)	(0.47)	(0.24)	(0.19)	(0.10)
SPAR WEBS						
BULKHEAD	0.34	0.36	0.28	0.10	0.19	0.30
TRUSS	0.30	0.54	0.49	0.19	—	—
Σ	(0.64)	(0.90)	(0.77)	(0.29)	(0.19)	(0.30)
RIB CAPS						
UPPER	0.06	0.08	0.12	0.08	0.08	0.08
LOWER	0.07	0.09	0.09	0.07	0.07	0.09
Σ	(0.13)	(0.17)	(0.21)	(0.15)	(0.15)	(0.17)
SPAR CAPS						
UPPER	0.24	1.23	1.22	0.69	0.12	0.16
LOWER	0.41	1.71	1.61	0.73	0.12	0.14
Σ	(0.65)	(2.94)	(2.83)	(1.42)	(0.24)	(0.30)
NON-OPTIMUM						
MECH. FAST.	0.18	0.20	0.20	0.05	0.03	0.04
WEB INTERS.	0.12	0.12	0.12	0.05	0.03	0.04
Σ	(0.30)	(0.32)	(0.32)	(0.10)	(0.06)	(0.08)
Σ POINT DESIGN WEIGHT $\frac{LB}{FT^2}$	3.80	6.79	6.99	4.60	7.37	7.44

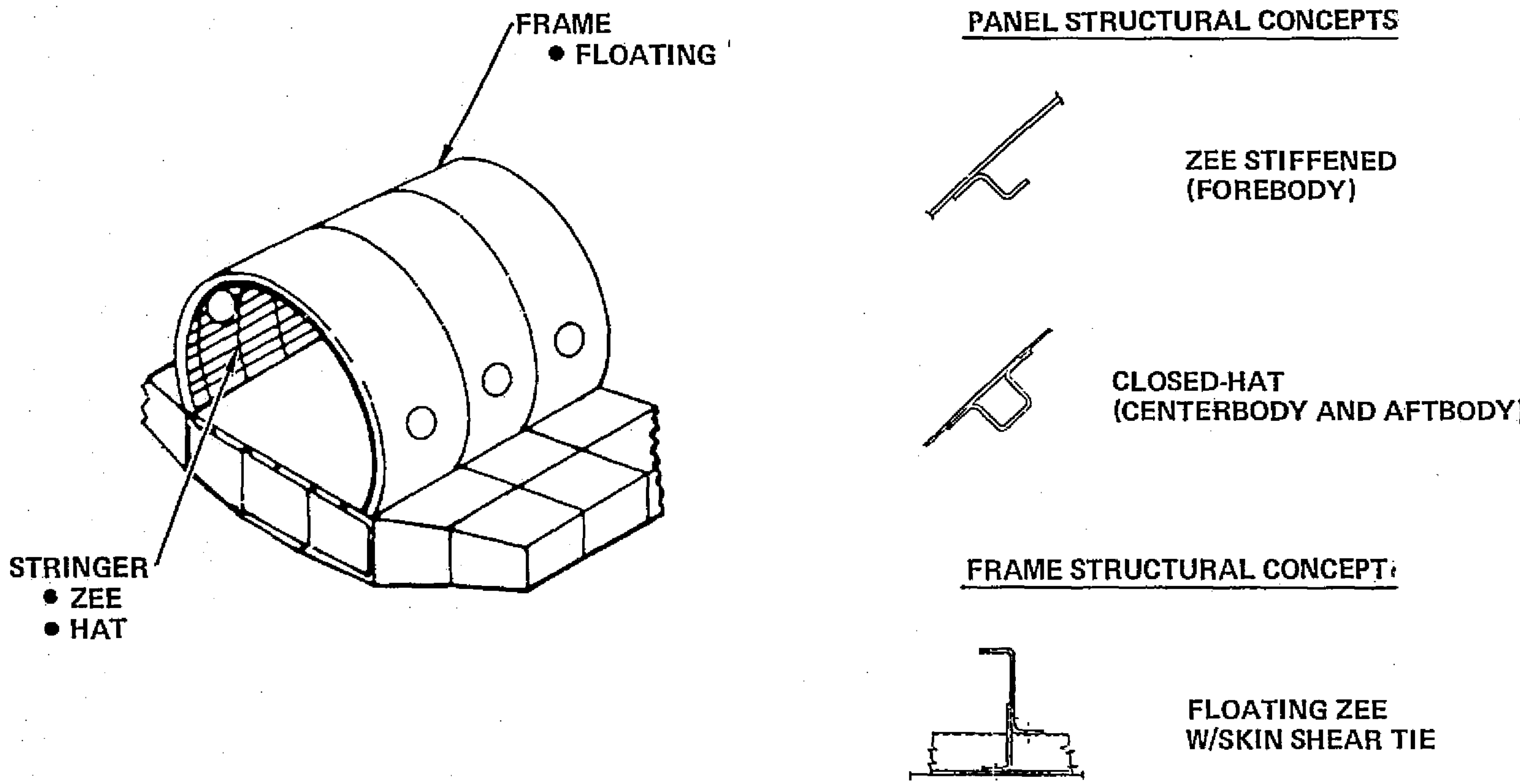


Figure 12-80. Fuselage Structural Approach For Task II

For the near-term conventional design, titanium alloy Ti-6Al-4V (annealed) material was used for the fuselage structural arrangement, i.e., panel and frame concepts. In addition to the conventional design, the potential weight savings associated with using composite reinforced panels was investigated. For this study Graphite/polyimide, Boron/polyimide, and Boron/aluminum material systems were investigated for reinforcing the crown of the metallic hat-stiffened panel concept.

Fuselage Point Design Regions

The four point design regions selected for analysis were FS 900, FS 1910, FS 2525, and FS 2900. These regions which are representative of the three general regions of a fuselage are shown in Figure 12-81 and includes one region on the fuselage forebody, two regions on the fuselage midbody (wing/fuselage interface), and an aftbody region.

In addition to the planform view, Figure 12-81 contains cross-sections indicative of the modeling technique employed to represent the frames in the 3-D structural model. The panel and frame element identification for these regions are shown in Figure 12-82. This identification system is used throughout the panel and frame analyses and is identical to that used to specify the elements in the 3-D structural model.

Fuselage Panel Analysis

The fuselage panel concepts were analyzed using both the conventional and composite reinforced material systems. This analysis was conducted at the four point design regions using the most critical point design environment for each region.

The results of this analysis are presented in the following text and included: (1) a section describing the methods of analysis, (2) the results of the metallic panel analysis, and (3) the results of the composite reinforced panel analysis.

<u>Fuselage Panel Method</u> - The fuselage panels were analyzed to determine the minimum weight design for each of the structural-material concepts and are discussed in the following paragraphs entitled: (1) panel loading, (2) stress analysis, and (3) allowables stress levels.

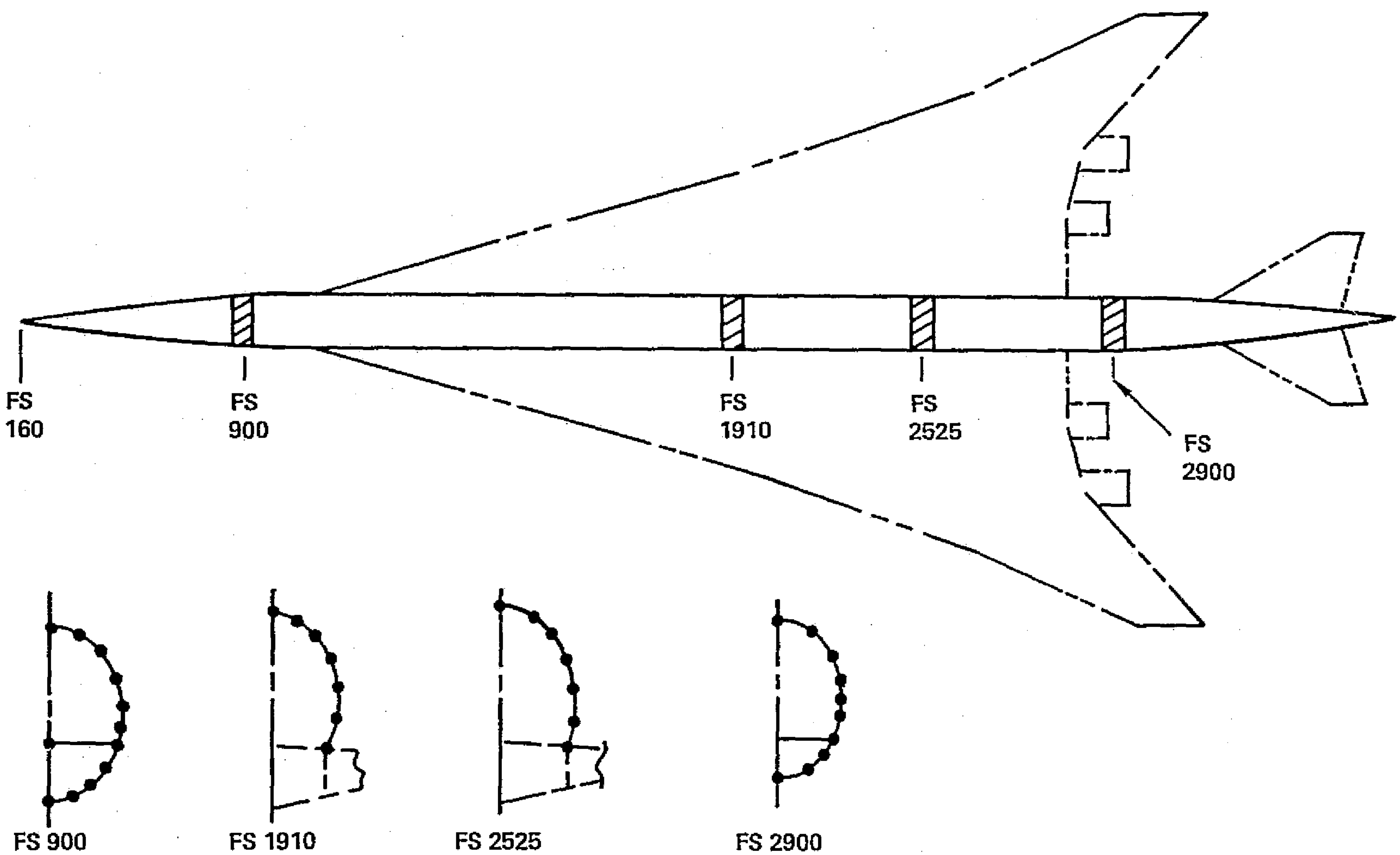


Figure 12-81. Definition of Fuselage Point Design Regions - Task II

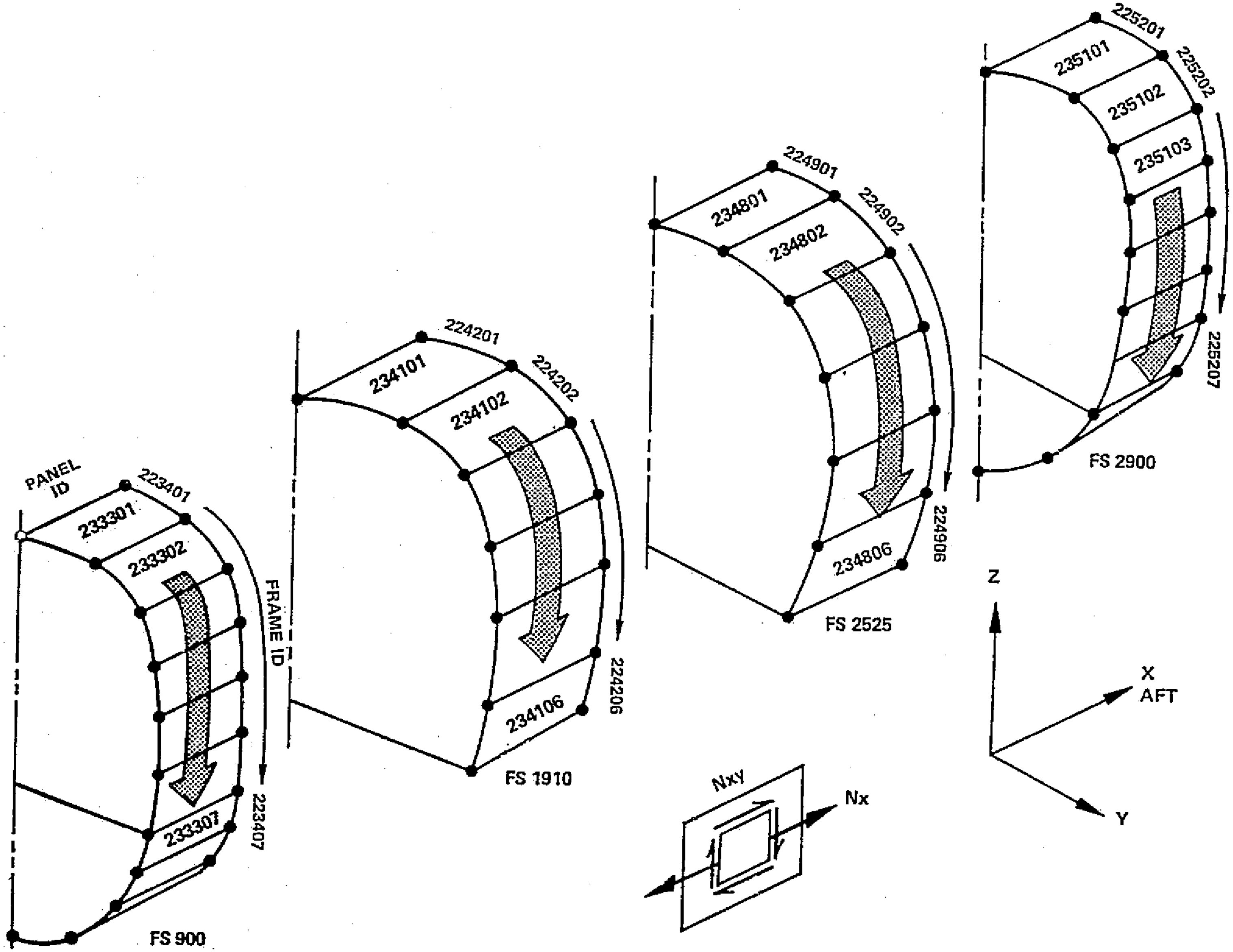


Figure 12-82. Fuselage Panel and Frame Identification - Task IIB

Panel Loading - The total inplane loads acting on the stiffened panels were defined by the point design environment and includes:

$$N_x = N_{x,air} + N_{x,thr}$$

$$N_{xy} = N_{xy,air} + N_{xy,thr}$$

$$N_y = N_{y,air}$$

where $N_{y,air}$ is equal to the hoop pressure force ($N_\theta = pr$), and $N_{x,air}$, $N_{x,thr}$, $N_{xy,air}$, and $N_{xy,thr}$ are the air load and thermal load intensities as derived from the NASTRAN redundant Structural Analysis using the 3-D structural model. The meridional pressure force N_ϕ was conservatively neglected when the air load component (N_x) was compression.

The Point Design Environment for all the flight conditions investigated are defined in Section 11 with the loads for the most critical flight condition repeated in Table 12-93.

Stress Analysis - The applied stresses on the panel are:

$$f_x = \frac{N_x}{\bar{t}}$$

$$f_{xy} = \frac{N_{xy}}{t}$$

$$f_y = \frac{N_y}{t}$$

Where $\bar{t}$ is the extensional thickness of the panel in the X-direction and t is the effective shear and membrane thickness.

TABLE 12-93. FUSELAGE POINT DESIGN ENVIRONMENT - TASK IIB, MACH 2.7 START-OF-CRUISE CONDITION

CONDITION 20 SYMMETRIC MANEUVER AT MACH 2.70 (START-OF-CRUISE), WEIGHT = 660,000 LB., n_z = 2.5

ITEM	UNITS	FS 900 (23XXXX)									FS 1910 (23XXXX)					
		3301	3302	3303	3304	3305	3306	3307	3308	3309	4101	4102	4103	4104	4105	4106
N_X	LB/IN	-56	-37	-8	-5	-37	-2	-13	-30	+72	-8618	-6284	-3779	-1885	-550	+611
N_{XY}	LB/IN	–	-7	-27	-44	-45	-27	-44	-7	+8	+358	+890	+1110	+1136	+1088	+1025
N_X, TH	LB/IN	-22	-9	+12	+25	+26	+29	+16	+1	-11	+165	-11	-255	-498	-738	-23
N_{XT}, TH	LB/IN	–	+4	+7	+4	+1	–	-9	-7	-10	-13	-32	-40	-42	-37	-27
AERO PRESS.	PSI	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
INTERNAL PRESS.	PSI	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55
NET PRESS.	PSI	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55
$T_{AVG.}$	°F	342	339	336	334	332	333	333	333	333	295	300	305	312	320	324
ΔT	°F	-105	-105	-106	-106	-106	-106	-106	-106	-106	-175	-170	-166	-162	-159	-157

ITEM	UNITS	FS 2525 (23XXXX)						FS 2900 (23XXXX)								
		4801	4802	4803	4804	4805	4806	5101	5102	5103	5104	5105	5106	5107	5108	5109
N_X	LB/IN	-12413	-7932	-4066	-1222	+664	+2319	-10441	-7108	-3674	-1071	+785	+2403	+4822	+7934	+11862
N_{XY}	LB/IN	+67	+67	+6	-59	-120	-187	-110	-368	-564	-676	-695	-697	-827	-387	-338
N_X, TH	LB/IN	+329	+57	-301	-646	-785	-272	+177	+37	-124	-261	-301	-375	-501	-220	+518
N_{XT}, TH	LB/IN	+6	+14	+26	+24	+14	-10	+10	+21	+17	+7	-8	-22	+16	-99	-8
AERO PRESS.	PSI	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
INTERNAL PRESS.	PSI	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.56	17.55	17.55	17.55	17.55	17.36	17.55
NET PRESS.	PSI	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55	17.55
$T_{AVG.}$	°F	281	287	293	300	307	311	292	295	298	300	301	295	288	283	278
ΔT	°F	-186	-18[illegible]	-180	-177	-173	-171	-174	-165	-156	-149	-147	-150	-157	-167	-177

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The combined stresses were calculated using the following equations. For compression, the Octahedral Shear Stress Theory was used to calculate the equivalent stress

$$f_{eq} = \left(f_x^{\,2} - f_x f_y + f_y^{\,2} + 3f_{xy}^{\,2}\right)^{1/2}$$

whereas, the combined tensile stresses were calculated using an equivalent stress of:

$$f_{eq} = \left(f_x^{\,2} + f_x f_y + f_y^{\,2} + f_{xy}^{\,2}\right)^{1/2}$$

Allowable Stresses - For the compression stress state both column buckling and crippling were considered. For a simply supported beam, the wide column theory for compressive buckling

$$N_{x,cr} = \frac{\pi^2 D_2}{L^2}$$

where

$$D_2 = \overline{\eta}\, E\, \overline{I}$$

in which

$\overline{\eta}$ = plasticity correction factor

$\overline{I}$ = area moment of inertia per unit width

The crippling strength of the stiffener and effective skin were determined using the theory and method presented in Reference 10. In this reference, the crippling stress is calculated for the stiffener (hat-section) by dividing the shape into its component flat and curved elements. Using these elements the crippling strength for each element is obtained and the average crippling strength of the section is determined.

$$F_{cc}(avg) = \frac{\Sigma F_{ccn}\, An}{\Sigma\, An}$$

For the tension condition, the ultimate design gross area stress is limited to 90,000 psi for symmetrical flight and ground conditions. Section 13 contains the fatigue analyses conducted to establish this design stress level.

Using the theory previously discussed, allowable curves were generated to facilitate the fuselage structural analyses. An example of the fuselage allowable loads are presented in Figure 12-83.

Metallic Panels - The conventional fuselage panels, Ti6Al-4V (ann.) material, were analyzed using the internal loads shown in Table 12-93 with the methods previously discussed. Table 12-94 contains the results of this analysis at each of the four point design regions.

The forward point design region at FS 900 is design by the operating condition, i.e., applied 1-g stresses at the mid-cruise condition compared to an operating design allowable stress of 25,000 psi. This condition resulted in a 0.036-inch thick zee-stiffener and skin being the least-weight concept.

The remaining three point design regions are designed for the design ultimate loads for the start-of-cruise condition. The closed hat-stiffened concept was analyzed keeping a constant 6.0 inch pitch, a crown width of 1.5 inches, and a height of 1.25 inches. Fixing these dimensions facilitates splicing and allows the use of a standard shear tie with only the thickness variable. The results of the analysis using these constraints are shown in Table 12-94 with the panel locations being defined in Figure 12-82. With reference to point design region FS 1910, the skin thickness t_s varied from 0.04- to 0.07-inch with the stringer thickness ranging from 0.03- to 0.06-inch. The equivalent panel thickness $\bar{t}$ ranged from a minimum of 0.069-inch on the side panel (234104) to a maximum thickness of 0.129-inch for the upper panel. Region FS 2525 has skin thicknesses ranging from 0.04-inch to 0.07-inch and stringer thicknesses varying from 0.03-inch to 0.08 inch. A maximum $\bar{t}$ of 0.149-inch occurs at the uppermost panel at FS 2525 and at the lowest panel at point design region FS 2900. For these four regions, the thicknesses ranged from 0.04-inch to 0.07-inch and from 0.03-inch to 0.08-inch for the skins and stringers, respectively.

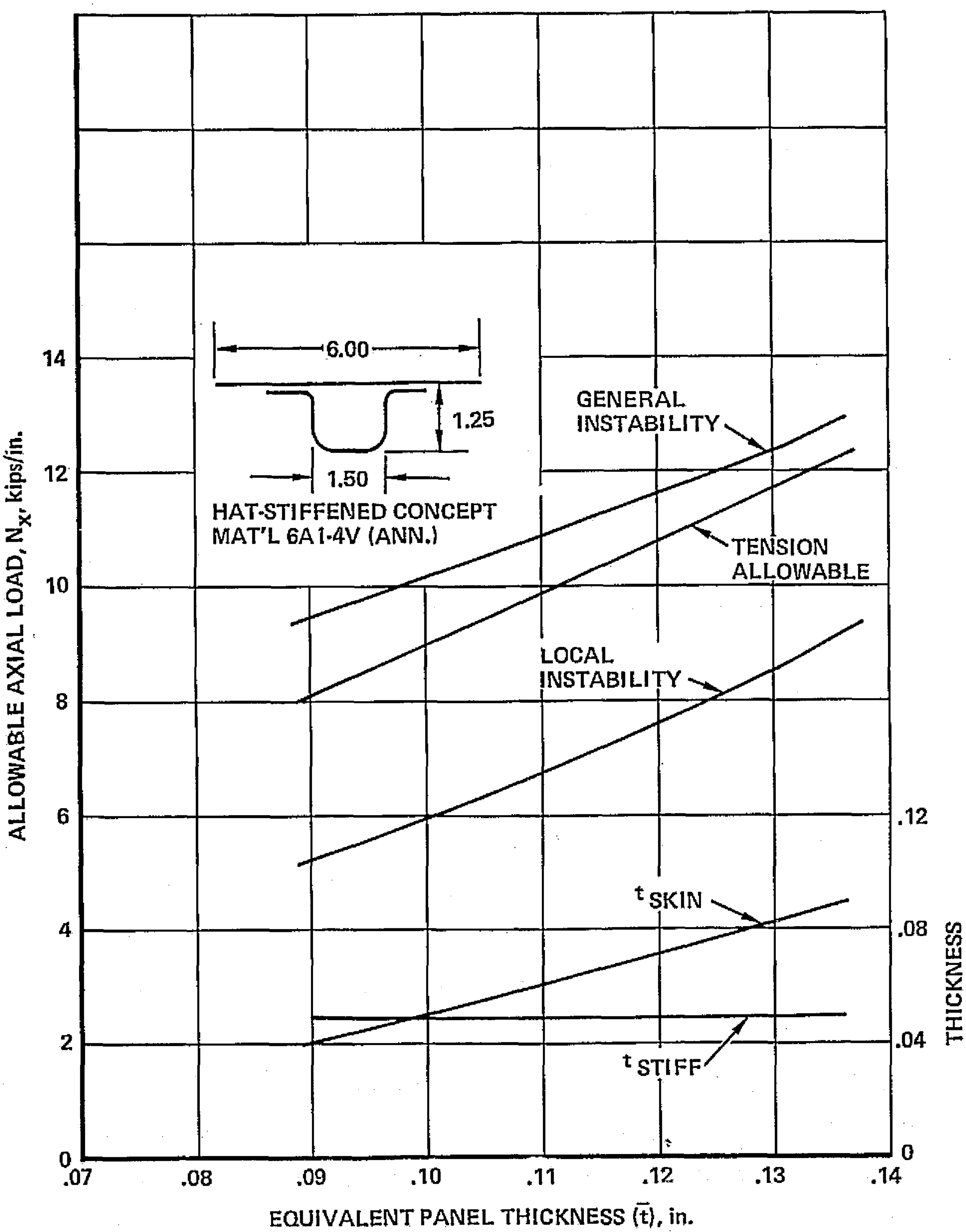
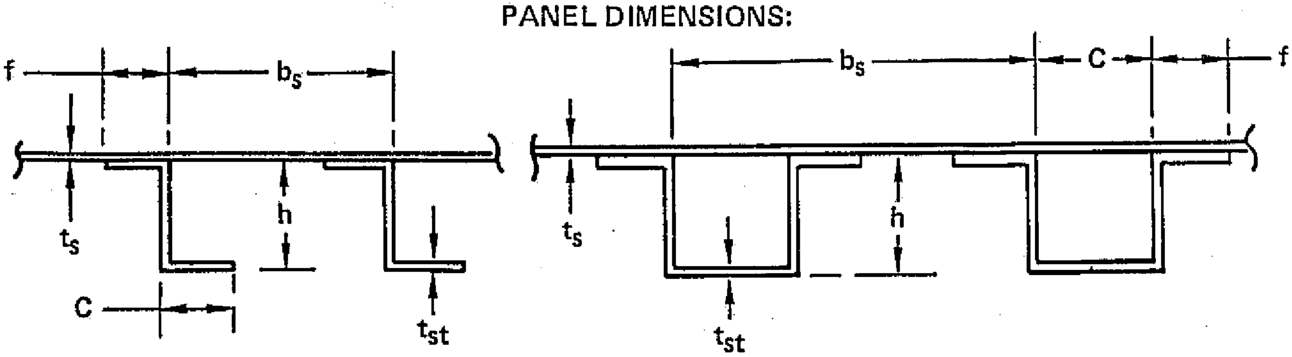


Figure 12-83. Fuselage Panel Allowable

TABLE 12-94. FUSELAGE PANEL GEOMETRY - TASK II

POINT DESIGN REGION	PANEL CONCEPT	CIRCUMF. LOCATION	FUSELAGE PANEL DIMENSIONS						
			b_s (in.)	t_s (in.)	C (in.)	f (in.)	h (in.)	t_{st} (in.)	$\bar{t}$ (in.)
FS 900	ZEE-STIFFENED	233301-233307	4.0	.036	.55	0.75	1.00	.036	.056
FS 1910	HAT-STIFFENED	234101	6.0	.07	1.5	0.80	1.25	.06	.129
		234102	6.0	.06	1.5	0.80	1.25	.05	.109
		234103	6.0	.04	1.5	0.80	1.25	.04	.079
		234104	6.0	.04	1.5	0.80	1.25	.03	.069
		234105	6.0	.05	1.5	0.80	1.25	.05	.099
		234106	6.0	.06	1.5	0.80	1.25	.06	.119
FS 2525	HAT-STIFFENED	234801	6.0	.07	1.5	0.80	1.25	.08	.149
		234802	6.0	.06	1.5	0.80	1.25	.06	.119
		234803	6.0	.05	1.5	0.80	1.25	.05	.099
		234804	6.0	.04	1.5	0.80	1.25	.03	.069
		234805	6.0	.04	1.5	0.80	1.25	.03	.069
		234806	6.0	.04	1.5	0.80	1.25	.04	.079
FS 2900	HAT-STIFFENED	235101	6.0	.07	1.5	0.80	1.25	.07	.139
		235102	6.0	.05	1.5	0.80	1.25	.06	.109
		235103	6.0	.05	1.5	0.80	1.25	.04	.089
		235104	6.0	.04	1.5	0.80	1.25	.03	.069
		235105	6.0	.04	1.5	0.80	1.25	.03	.069
		235106	6.0	.04	1.5	0.80	1.25	.03	.069
		235107	6.0	.05	1.5	0.80	1.25	.04	.089
		235108	6.0	.05	1.5	0.80	1.25	.06	.109
		235109	6.0	.07	1.5	0.80	1.25	.08	.149

PANEL DIMENSIONS:



ZEE-STIFFENED CONCEPT

HAT-STIFFENED CONCEPT

The average panel thickness ($\bar{t}$) and unit weight (w) were calculated at each of the four point design regions. An example of the technique used to calculate these average thicknesses is shown in Table 12-95 for FS 2565. These results are summarized in Table 12-96 for each of the regions.

A maximum value of approximately 2.5 lb/sq.ft is noted for regions FS 2565 and FS 2900, with the forebody region (FS 900) have a minimum-weight of approximately 1.3 lb/sq.ft. A value of 2.40 lb/sq.ft is noted for region FS 1910.

Composite Reinforced Panels - An initial trade-off study was performed to assess the merits of reinforcing the crown of the hat-stiffener with Graphite/polyimide, Boron/polyimide, and Boron/aluminum composites. This study was conducted on the uppermost panels (maximum compressive loaded panel) at FS 1910, FS 2525, and FS 2900. The crown reinforcement and the metal hat were optimized for the applied compressive loads except for the constraint of a 6.0 inch stringer pitch. Skin failure was limited to the initial buckling strength.

The results of this study are shown in Table 12-97 and indicate for these high compressive loaded panels the Boron/polyimide reinforcement affords the least-weight design at each of the three point design regions. In general, the Graphite/polyimide and Boron/aluminum designs had almost equal weights which are approximately 0.05 lb/sq.ft heavier than the least-weight Boron/polyimide design. The Boron/polyimide design at FS 2525 exhibits the largest percentage weight savings over the homogeneous metal design, approximately 16-percent.

The second step in the composite reinforced study was a more detailed investigation of the lightest-weight reinforcement (Boron/polyimide) determined from the initial trade-off study. Table 12-98 summarizes the results of this investigation. This analysis was conducted at FS 2900 using Boron/polyimide reinforced stringers at each circumferential panel locations and included both constrained and non-constrained geometries. For the non-constrained geometry designs, only the pitch was held constant at 6.0-inches, the average unit weight of the panels at FS 2900 was 2.21 lb/sq.ft. The corresponding weight for the constrained Boron/polyimide reinforced hat-stiffened panel was 2.35 lb/sq.ft. These weights when compared to the all titanium design, unit weight of 2.56 lb/sq.ft, indicate a weight savings of approximately 8-percent and 14-percent for the constrained and unconstrained Boron/polyimide reinforced design, respectively.

TABLE 12-95. AVERAGE PANEL THICKNESS FOR FS2565 - TASK IIB

POINT DESIGN REGION	PANEL CONCEPT	PANEL ELEMENT	$\bar{t}_i$ (in.^{2}/in.)	C_i (in.)
FS 2565	HAT-STIFFENED	234801	0.149	39.64
		234802	0.119	29.72
		234803	0.099	23.68
		234804	0.069	17.86
		234805	0.069	11.82
		234806	0.079	11.90
	AVG. VALUES	$\bar{t}$ = .1098 in.^{2}/in.; W = 2.53 lb./sq.ft.		

$$\bar{t} = \sum_{i=1}^{6} c_i \bar{t}_i \sum_{i=1}^{6} c_i$$

$$W = 23.04 \times \bar{t}$$

TABLE 12-96. FUSELAGE PANEL WEIGHTS - TASK IIB

POINT DESIGN REGION	PANEL CONCEPT	$\bar{t}$ (in.^{2}/in.)	w (lb./sq.ft)
FS 900	ZEE-STIFFENED	0.056	1.29
FS 1910	HAT-STIFFENED	0.104	2.40
FS 2565	HAT-STIFFENED	0.110	2.53
FS 2900	HAT-STIFFENED	0.111	2.56

$\bar{t}$ = AVERAGE EQUIVALENT PANEL THICKNESS

w = AVERAGE PANEL UNIT WEIGHT

TABLE 12-97. WEIGHT COMPARISON OF COMPOSITE REINFORCED PANEL CONCEPTS

POINT DESIGN REGION	UNIT WEIGHT (w), lb/sq.ft. AND PERCENTAGE WEIGHT SAVING(1)			
	STRINGER REINFORCEMENT MAT'L			
	NONE (ALL TITANIUM)	GR/PI	B/PI	B/AI
FS 1910	2.97	2.80	2.73	2.77
	–	5.7%	8.1%	7.2%
FS 2525	3.44	2.94	2.90	2.96
	–	14.5%	15.7%	13.9%
FS 2900	3.20	2.83	2.77	2.83
	–	11.6%	13.4%	11.6%

NOTE:

1. PERCENTAGE WT. SAVING OVER THE ALL METALLIC DESIGN.

TABLE 12-98. WEIGHT COMPARISON OF B/PI REINFORCED PANEL CONCEPTS

POINT DESIGN REGION	PANEL CONCEPT	PANEL		UNIT WEIGHT (2), lb./sq.ft.		
				STRINGER DESIGN		
		ID	WIDTH (in.)	ALL TITANIUM (CONSTRAINED)	B/PI REINF. (OPTIMUM)	B/PI REINF. (CONSTRAINED)
FS 2900	HAT-STIFFENED	235101	39.70	3.20	2.77	2.78
		235102	28.36	2.51	2.13	2.46
		235103	22.70	2.05	1.86	2.01
		235104	17.00	1.59	1.42	1.53
		235105	11.36	1.59	1.42	1.53
		235106	11.32	1.59	1.42	1.53
		235107	17.04	2.05	1.86	2.01
		235108	17.00	2.51	2.19	2.36
		235109	39.71	3.43	2.85	2.98
AVG. w, lb./sq.ft. = $\sum_{i=1} C_i w_i \Big/ \sum_{i=1}^{n} C_i$ =				2.56	2.21	2.35
AVG. % WT. SAVING				–	13.7	8.20

Fuselage Frame Analysis

The fuselage circumferential frame elements at each point design region were analyzed to define the frame weight-strength relationship. The typical frame construction is shown in Figure 12-84 and includes views of the frame construction between panel stiffeners and at the stiffeners. The typical frame is of sheet metal construction and is 4.0 inches deep with a flange width of 0.75 inch. The frames are constructed to allow a 1.25-inch opening for the stringer run-through with skin shear ties provided between stiffeners.

The methods used for analyzing the frames are outlined in Figure 12-85. This figure presents the approach which includes the frame section properties, applied stress equation, and the frame allowable stresses. The applied frame loads (axial force, transverse shear, and bending moment) were obtained from the results of the NASTRAN redundant structure analysis solution using the 3-D structural model. An example of these model frame loads are presented in Figures 12-86 and 12-87 for point design region FS 2525. These model frame loads reflect the maximum frame bending moments and corresponding axial loads for the positive and negative gust conditions, conditions 23 and 24, respectively.

Since the structure model reflect lumped section properties (i.e., one model frame represents more than one actual frame) the model loads have to be reduced to reflect a unit frame. Figure 12-88 presents the relationship between actual frames and model frames with the model frame at FS 3000 having the highest ratio (ten-to-one) of the four point design regions. The midbody regions (FS 1955 and FS 2565) and the forebody regions (FS 1000) have ratios of four-to-one and seven-to-one, respectively. Generally the actual frames lumped into any specific model frame represent a small enough region to conclude that each frame has approximately equal axial and bending stiffness and a linear delumping of model loads (model loads divided by number of actual frames) can be conducted without significant error. Frame stress analyses were conducted at each of the four point design regions using the method outlined in Figure 12-85 and the applied loads determined from the 3-D structural model. An example of this analysis is presented in Table 12-99 for point design region FS 2565. With reference to this table, the frame element identification number coincides with the identification system used for the 3-D structural model. In addition, the number of actual frames lumped into the model frame are specified.

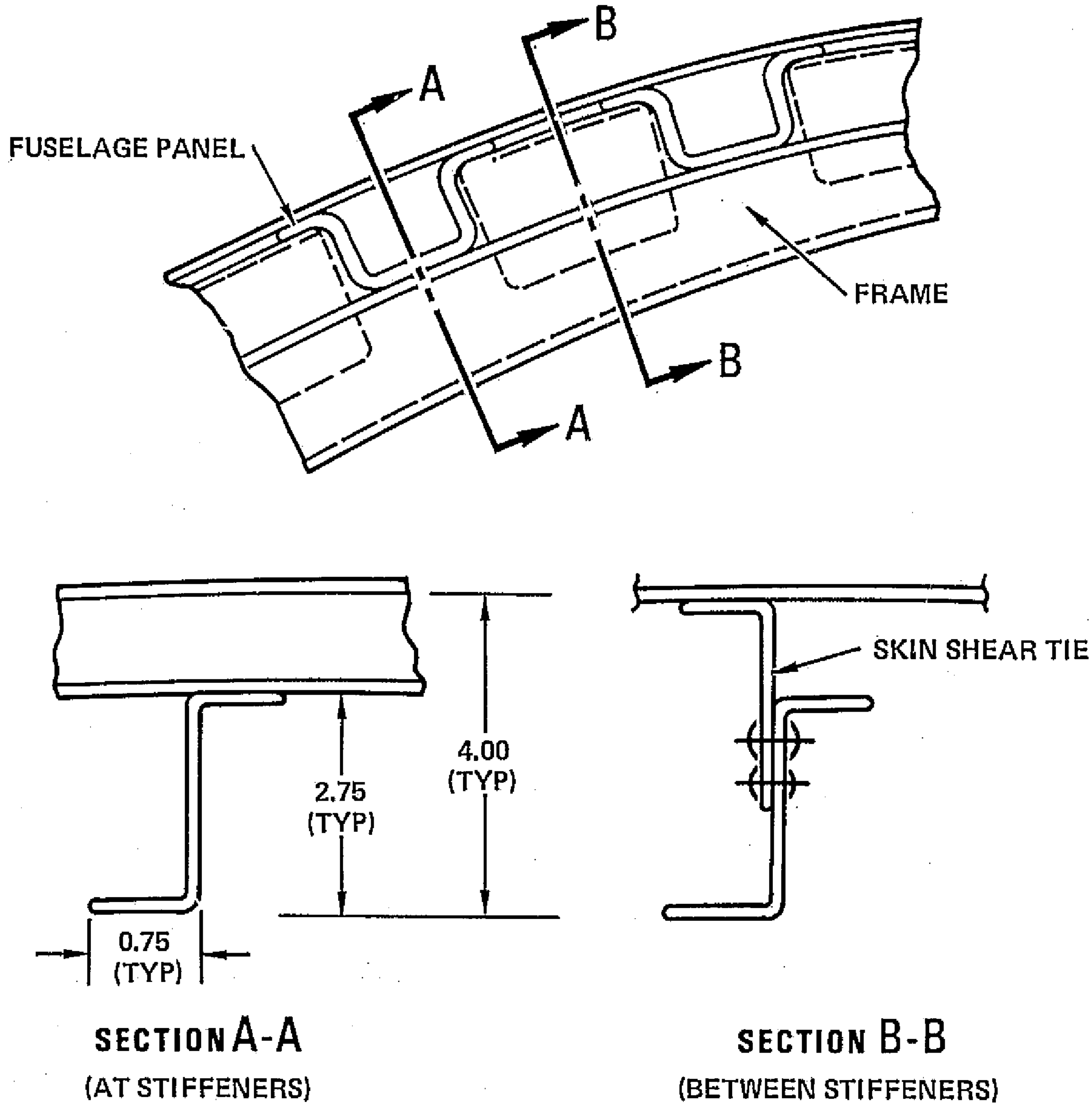


Figure 12-84. Fuselage Frame Geometry

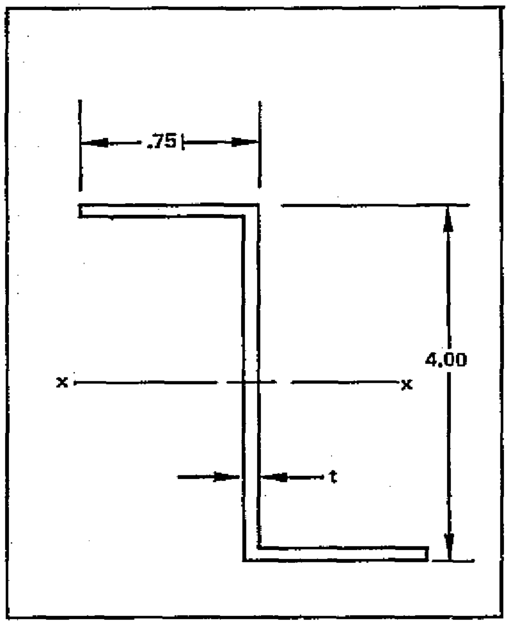
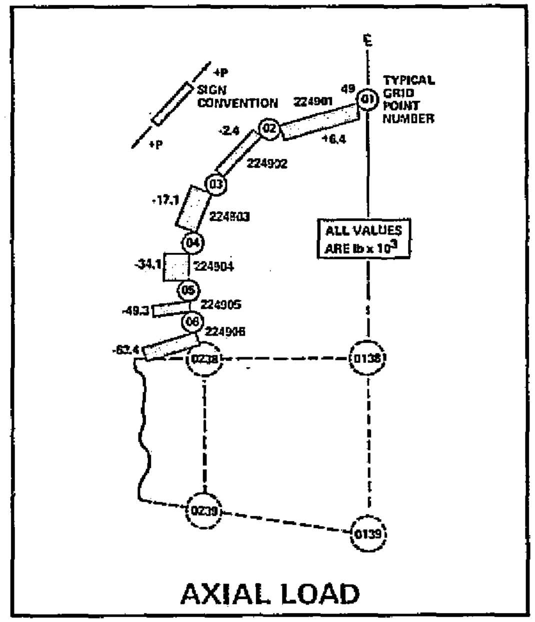


FIGURE 12-85a
FRAME CROSS SECTION



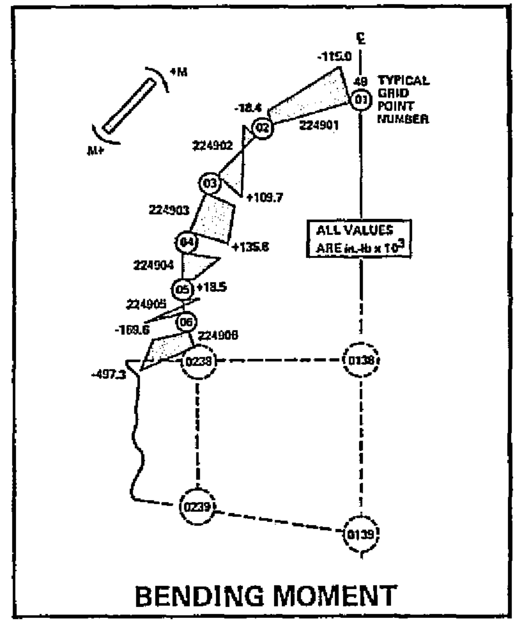


FIGURE 12-85b FRAME LOADS

- SECTION PROPERTIES PER FIGURE 12-85a

  $A = 5.50t \text{ in.}^2;\ C = 2.00 \text{ in.};\ I = 11.33\, t \text{ in.}^4$

- APPLIED STRESSES:

  $$f = \frac{MC}{I} \pm \frac{P}{A} = 0.176\left(\frac{M}{t}\right) \pm 0.182\left(\frac{P}{t}\right)$$

  WHERE:

  P = FRAME AXIAL LOAD } PER NASTRAN STATIC
  M= FRAME BENDING MOMENT } SOLUTION, FIGURE 12-85b

- ALLOWABLE STRESSES PER FIGURE 12-85c
  COMPRESSION - LOCAL CRIPPLING
  TENSION - FATIGUE CUT-OFF STRESS (90,000 psi)

FIGURE 12-85a

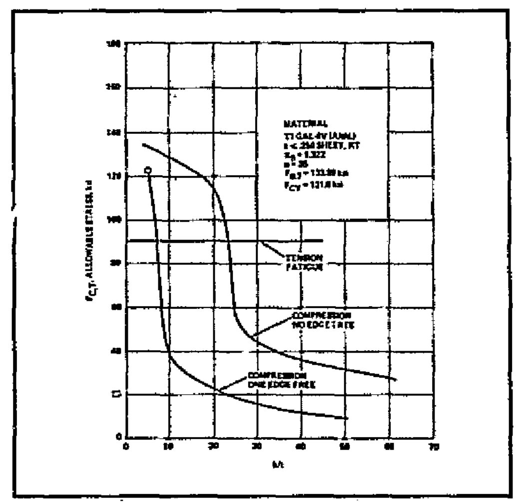
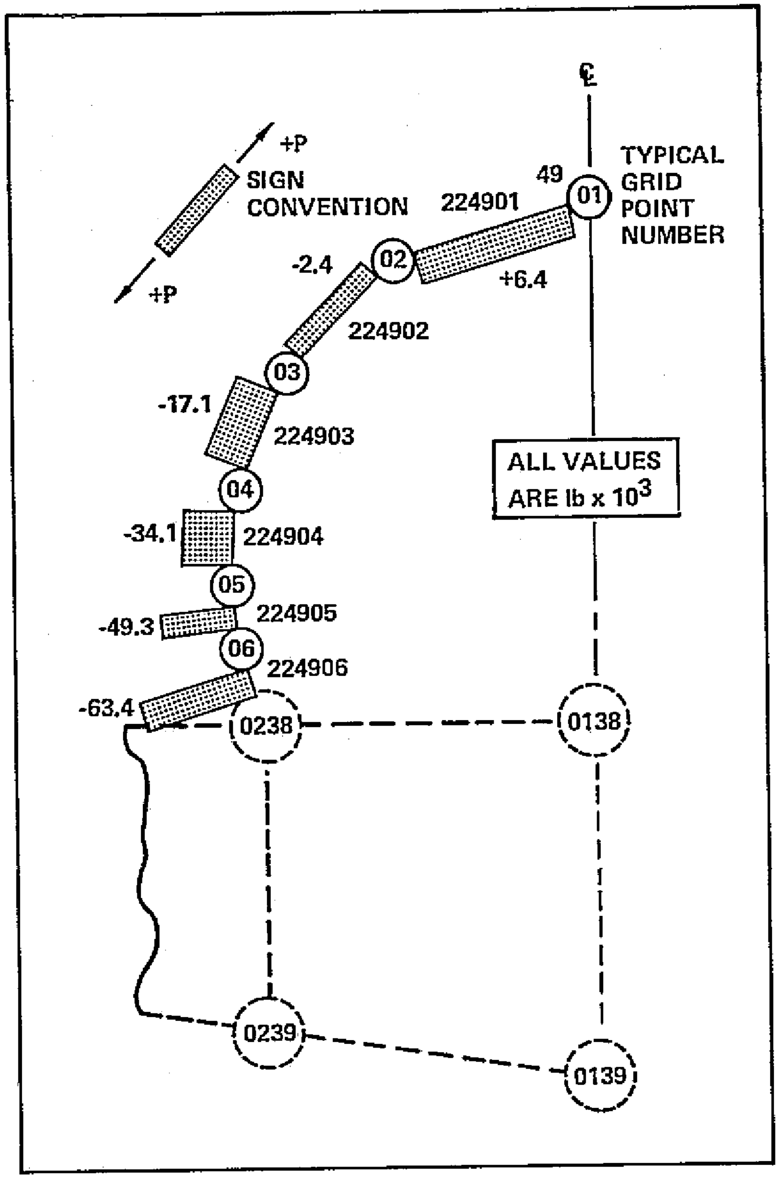
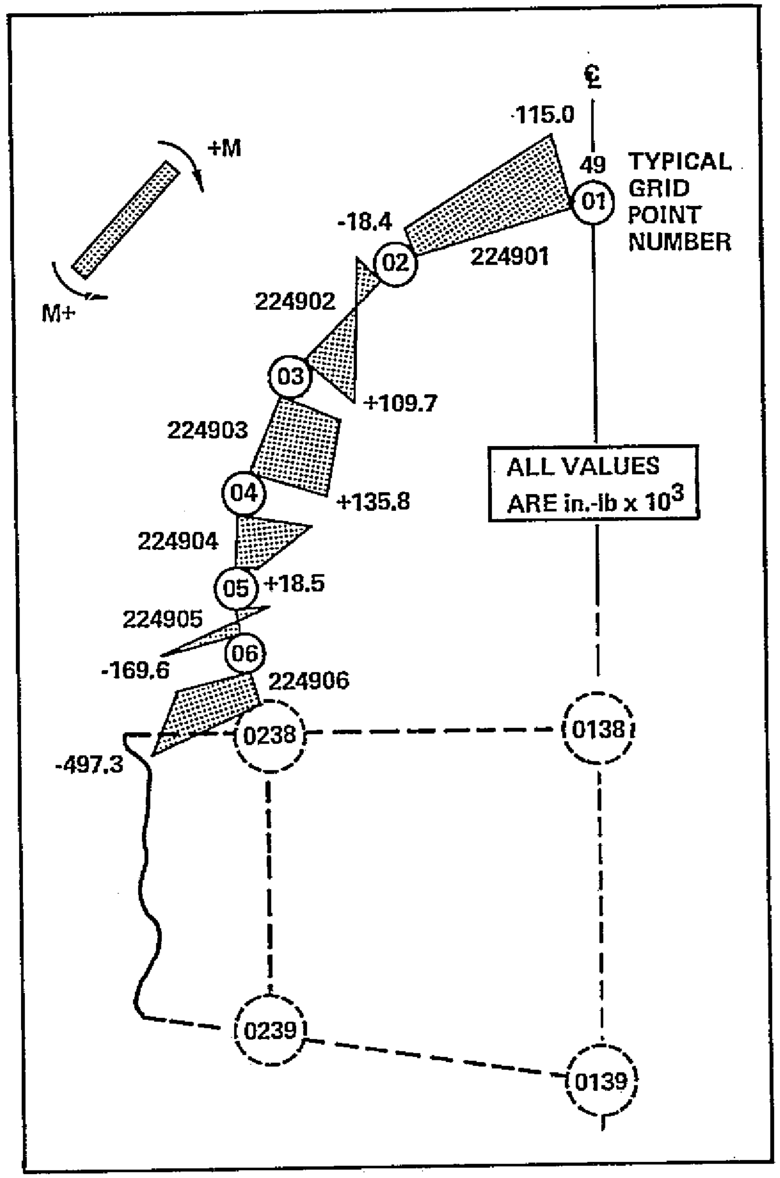


FIGURE 12-85c ALLOWABLES

Figure 12-85. Fuselage Frame Method of Analysis

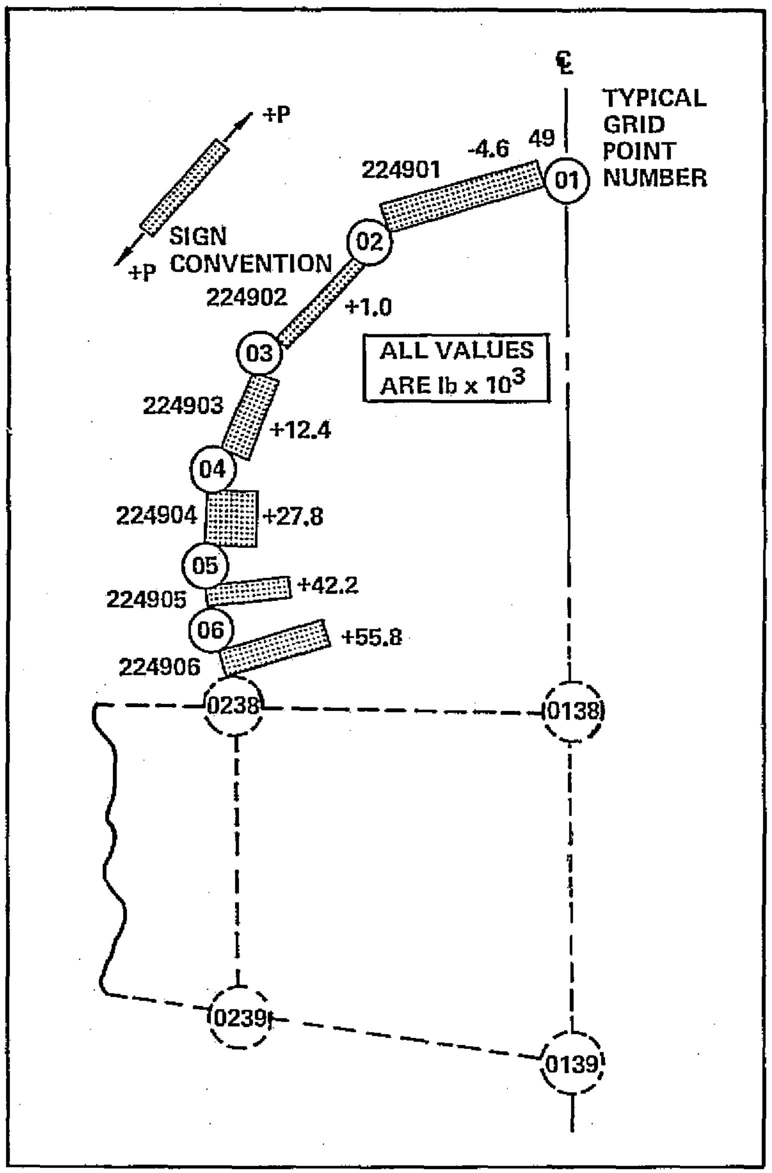


AXIAL LOAD
FIGURE 12-86a

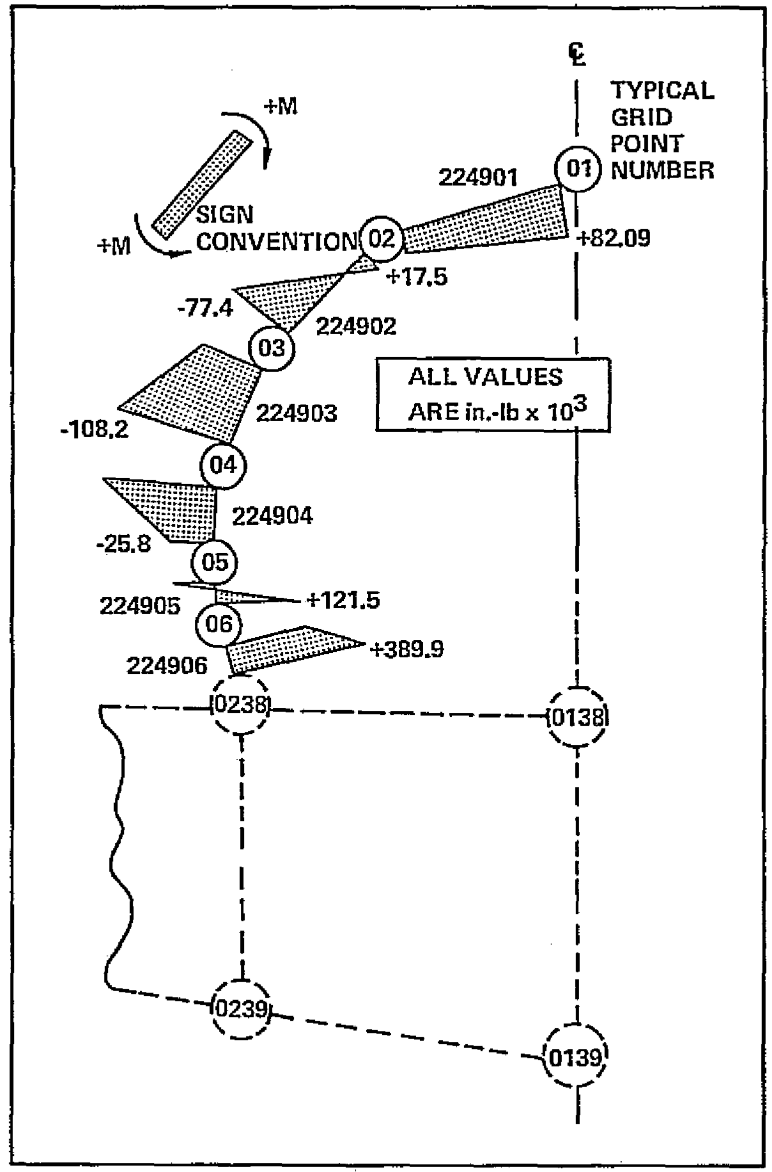


BENDING MOMENT
FIGURE 12-86b

Figure 12-86. Model Frame Loads for FS 2565, Condition 23, Static Gust (Positive) at Mach 0.90

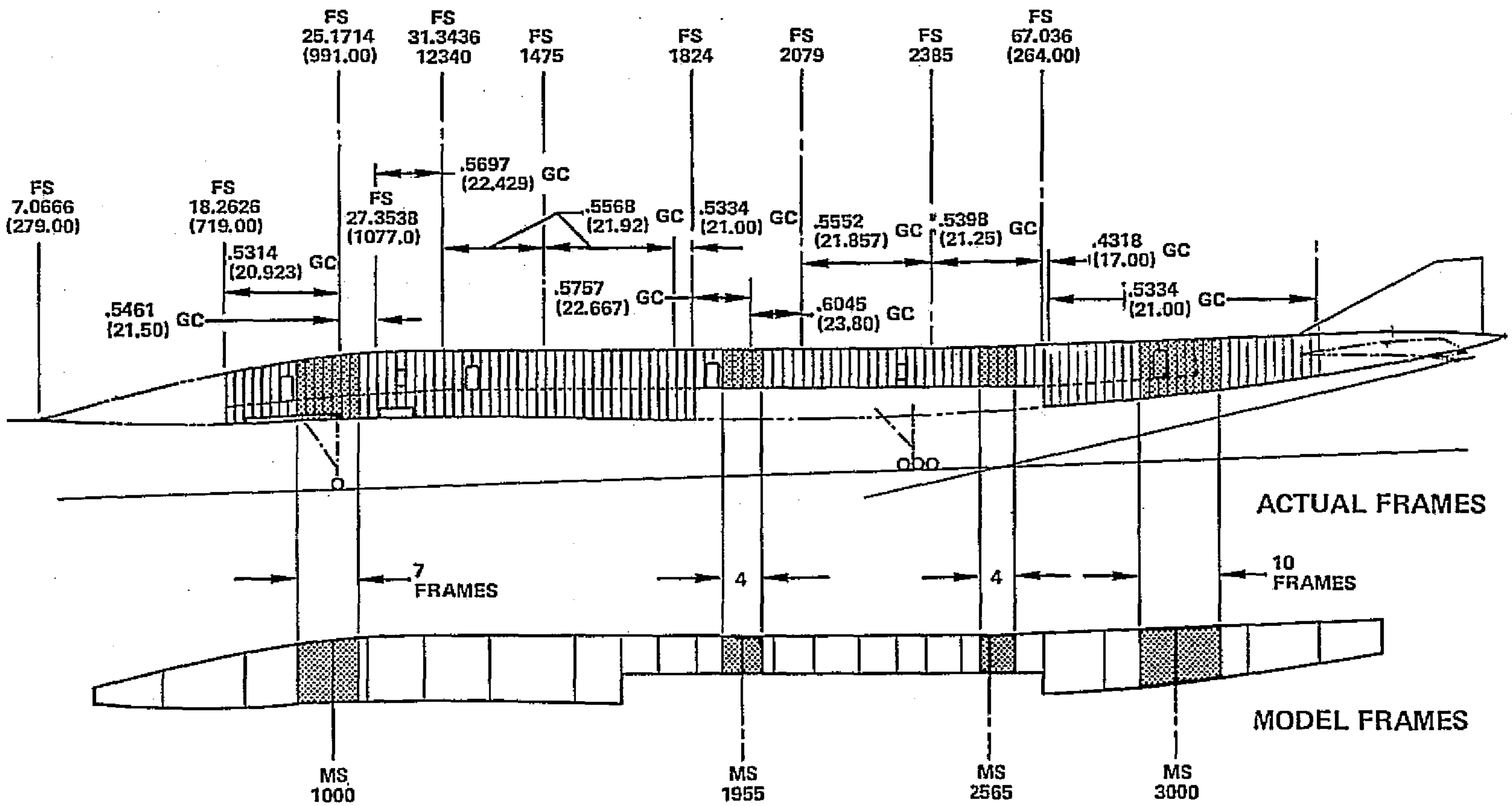


AXIAL LOAD
FIGURE 12-87a



BENDING MOMENT
FIGURE 12-87b

Figure 12-87. Model Frame Loads for FS 2565, Condition 24, Static Gust (Negative) at Mach 0.90



LUMPED FRAMES	7	4	4	10
AVG. FRAME SPACING	21.21	23.23	21.25	21.00

Figure 12-88. Lumped Model Frames

TABLE 12-99. FUSELAGE FRAME ANALYSIS AT POINT DESIGN REGION FS2565

FRAME ELEMENT	NO. ACT. FR. n	BEAM END	MAXIMUM NEGATIVE MOMENT CONDITION									MAXIMUM POSITIVE MOMENT CONDITION								
			COND. NO.	M_L x10^3 (in.-lb)	M x10^3 (in.-lb)	P_L x10^3 (lb)	P x10^3 (lb)	LOAD INTENSITY N_{IN} (lb/in.)	LOAD INTENSITY N_{OUT} (lb./in.)	REQUIRED THICKNESS t_{IN} (in.)	REQUIRED THICKNESS t_{OUT} (in.)	COND. NO.	M_L x10^3 (in.-lb)	M x10^3 (in.-lb)	P_L x10^3 (lb)	P x10^3 (lb)	LOAD INTENSITY N_{IN} (lb/in.)	LOAD INTENSITY N_{OUT} (lb/in.)	REQUIRED THICKNESS t_{IN} (in.)	REQUIRED THICKNESS t_{OUT} (in.)
224901	4	A B	23	-115.0 (-66.7) -18.4	-16.7	6.40	1.60	3,200	-2,600	.036	.073	24	82.9 (50.2) 17.5	12.5	-4.60	-1.15	-2,420	2,000	.069	(MIN) .030
224902	4	A B	24	+17.5 (-29.9) -77.4	-7.5	1.02	0.26	1,400	-1,300	(MIN) .030	.047	23	-18.4 (45.7) 109.7	11.4	-2.45	-0.61	-2,120	1,900	.065	(MIN) .030
224903	4	A B	24	-77.4 (-92.8) -108.2	-23.2	12.4	3.10	4,650	-3,520	.052	.083	23	109.7 (122.7) 135.8	30.7	-17.1	-4.27	-6,180	4,620	.096	.051
224904	4	A B	24	-108.2 (-66.9) -25.8	-16.7	27.8	6.90	4,210	-1,680	.047	.057	23	135.8 (77.2) 18.5	19.3	-34.1	-8.52	-4,950	1,840	.090	(MIN) .030
224905	4	A B	23	18.5 (-75.5) -169.6	-18.9	-49.3	-12.3	1,080	-5,560	(MIN) .030	.092	24	-25.8 (47.8) 121.5	12.0	42.2	10.6	-180	4,020	(MIN) .030	.045
224906	4	A B	23	-169.6 (-333.4) -497.3	-83.4	-63.4	-15.9	11,780	-17,560	.131	.146	24	121.5 (255.7) 389.9	63.9	55.8	13.9	-8,710	13,800	.107	.153

1. NOMENCLATURE

- M_L = MODEL LUMPED MOMENT
- M = MOMENT PER FRAME = M_L(AVG)/n
- P_L = MODEL LUMPED AXIAL LOAD
- P = AXIAL LOAD PER FRAME = P_L/n
- N_i = FLANGE LOAD INTENSITY (lb/in.)
- t_i = REQUIRED FLANGE THICKNESS

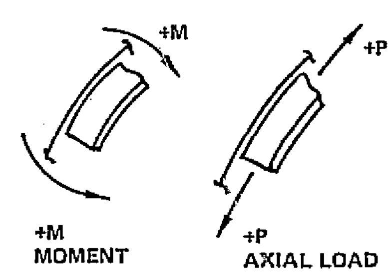
N_i = 0.182 P $\pm$0.176M

t_i = N_i/F

F = ALLOWABLE TENSION/COMPRESSION STRESS

n = NUMBER OF ACTUAL FRAMES

2. SIGN CONVENTION:



The identification system and frame lumping ratios are shown in Figures 12-82 and 12-88. Ends A and B refer to the ends of the bar element used in modeling the frame.

As a result of the redundant structural analysis the model frame loads are available for all flight conditions. These frame loads (axial load, transverse shear, and bending moment) are scanned to define the most critical loads for each frame circumferential element at the point design regions. For design region FS 2525 the maximum positive and negative bending moments (M_L) and the corresponding axial load (P_L) are listed for the critical flight conditions. These loads occur for the positive and negative gust flight conditions, conditions 23 and 24, respectively. In addition, the average moment of each element, which is used in the analysis, is displayed in parenthesis on the referenced table. The unit frame loads (M and P) are determined by dividing the model loads (M_L and P_L) by the number of actual frames. For the specified frame shown in Figure 12-85a the maximum fiber load intensities are calculated for the inner and outer flanges using the equation specified in Figure 12-85. The required frame thicknesses (t_{in} and t_{out}) are calculated for the tension and compression conditions using allowable curve similar to those specified in Figure 12-85. Having obtained these thicknesses, the maximum values noted on Table 12-99, define the final frame thickness.

The results of the point design stress analysis are used to calculate the equivalent panel thickness ($\bar{t}$) and unit weight (w) of the frames which are shown in Tables 12-100 and 12-101 for the four point design regions. These tables contain the individual frame element properties as well as the average equivalent panel thickness and unit weight. The nomenclature and equations for calculating these values are defined in the footnotes. For clarity, the results of these calculations are summarized in Table 12-102. A maximum unit weight of 0.51 lb/sq.ft is noted for point design region FS 2525 with regions FS 900 and FS 2900 having approximately equal values of 0.20 lb/sq.ft. A value of 0.46 lb/sq.ft is indicated for FS 1910.

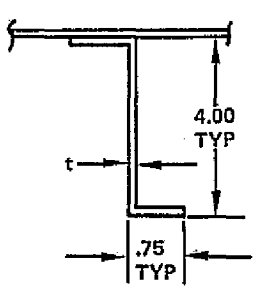
Fuselage Results

The results of the panel and frame analyses were combined to establish the weight trends of the major fuselage components. These values (sum of panel and frame weight) are used as the basis for extrapolating to the total fuselage weight as explained in the Mass Section, Section 15.

TABLE 12-100. FRAME GEOMETRY AND WEIGHT DATA FOR POINT DESIGN REGIONS FS900 AND FS1910

POINT DESIGN REGION	FRAME ELEMENT	FUSELAGE FRAME PROPERTIES				
		t (in.)	A (in.^{2})	b (in.)	$\bar{t}_i$ (in.^{2}/in.)	C_i (in.)
FS 900	223401	.039	.214	21.21	.0101	39.19
	223402	.030	.165	21.21	.0078	31.35
	223403	.047	.258	21.21	.0122	25.09
	223404	.039	.214	21.21	.0101	18.84
	223405	.030	.165	21.21	.0078	12.55
	223406	.045	.248	21.21	.0117	12.55
	223407	.041	.226	21.21	.0106	16.91
	223408	.030	.165	21.21	.0078	16.91
	223409	.032	.176	21.21	.0083	21.12
	223410	.030	.165	21.21	.0078	43.95
	AVG. VALUES	$\bar{t}$ = .0093 in.^{2}/in.; W = 0.21 lb./sq.ft.				
FS 1910	224201	.088	.484	23.23	.0208	39.64
	224202	.065	.358	23.23	.0154	29.72
	224203	.096	.528	23.23	.0227	23.68
	224204	.090	.495	23.23	.0213	17.86
	224205	.064	.352	23.23	.0152	11.82
	224206	.114	.627	23.23	.0270	11.90
	AVG. VALUES	$\bar{t}$ = .020 in.^{2}/in.; W = 0.46 lb./sq.ft.				

FRAME DIMENSIONS:



t = FRAME THICKNESS

A = FRAME AREA; 5.50 x t

b = FRAME SPACING

$\bar{t}_i$ = A/b

C_i = FRAME CIRCUMFERENCE

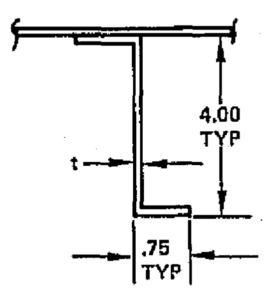
$$\bar{t}(\text{AVG.}) = \sum_{i=1}^{n} C_i \bar{t}_i \Big/ \sum_{i=1}^{n} C_i$$

$$W(\text{AVG.}) = 23.04 \times t(\text{AVG.})$$

TABLE 12-101. FRAME GEOMETRY AND WEIGHT DATA FOR POINT DESIGN REGIONS FS2525 AND FS2900

POINT DESIGN REGION	FRAME ELEMENT	FUSELAGE FRAME PROPERTIES				
		t (in.)	A (in.^{2})	b (in.)	$\overline{t}_i$ (in.2/in.)	C_i (in.)
FS 2525	224901	.073	.402	21.25	.0189	39.64
	224902	.065	.358	21.25	.0168	29.72
	224903	.096	.528	21.25	.0248	23.68
	224904	.090	.495	21.25	.0233	17.86
	224905	.092	.506	21.25	.0238	11.82
	224906	.153	.842	21.25	.0396	11.90
	AVG. VALUES	$\overline{t}$ = .0223 in.^{2}/in.; w = 0.51 lb./sq.ft.				
FS 2900	225201	.035	.192	21.00	.0092	39.70
	225202	.035	.192	21.00	.0092	28.36
	225203	.033	.182	21.00	.0086	22.70
	225204	.030	.165	21.00	.0079	17.00
	225205	.030	.165	21.00	.0079	11.36
	225206	.030	.165	21.00	.0079	11.32
	225207	.030	.165	21.00	.0079	17.04
	225208	.035	.192	21.00	.0092	17.00
	225209	.030	.165	21.00	.0079	39.71
	AVG. VALUES	$\overline{t}$ = .0085 in.^{2}/in.; w = 0.20 lb./sq.ft.				

FRAME DIMENSIONS:



t = FRAME THICKNESS

A = FRAME AREA; 5.50 x t

b = FRAME SPACING

$\overline{t}_i$ = A/b

C_i = FRAME CIRCUMFERENCE

$$\overline{t}(\text{AVG.}) = \sum_{i=1}^{n} C_i \overline{t}_i \Big/ \sum_{i=1}^{n} C_i$$

$$w(\text{AVG.}) = 2.304 \times \overline{t}(\text{AVG.})$$

The unit weights of the metallic fuselage design are summarized in Table 12-103. This table lists the unit weight of each component (panel and frame), and the combined total weight at each point design region. The heaviest-weight region is FS 2525 which has a total unit weight of 3.04 lb/sq.ft with the panel and frame components weighing 2.53 lb/sq.ft and 0.51 lb/sq.ft, respectively. Conversely, FS 900 has the least unit weight, 1.51 lb/sq.ft, which is composed of 1.29 lb/sq.ft. for the panel and 0.22 lb/sq.ft for the frame.

TABLE 12-102. SUMMARY OF FRAME GEOMETRY AND WEIGHT

POINT DESIGN REGION	FUSELAGE FRAME PROPERTIES			
	FRAME SPACING, b (in.)	AREA, A (in.^{2})	$\bar{t}$ (in.^{2}/in.)	w (lb./sq.ft.)
FS 900	21.21	0.197	.0093	.21
FS 1910	23.23	0.465	.0200	.46
FS 2525	21.25	0.474	.0223	.51
FS 2900	21.00	0.178	.0085	.20

A = AVERAGE FRAME AREA, in^2.

$$= \sum_{i=1}^{n} c_i A_i \Big/ \sum_{i=1}^{n} c_i$$

$\bar{t}$ = EQUIVALENT SURFACE PANEL AREA, in.^{2}/in.

= A/b

w = EQUIVALENT SURFACE PANEL WEIGHT, lb./sq. ft.

= 23.04 x $\bar{t}$

TABLE 12-103. DETAIL FUSELAGE WEIGHTS FOR THE TASK II STRUCTURAL ARRANGEMENT

POINT DESIGN REGION	FUSELAGE UNIT WEIGHTS (lb./sq.ft.)		
	PANEL WEIGHT	FRAME WEIGHT	TOTAL WEIGHT
FS 900	1.29	0.22	1.51
FS 1910	2.40	0.46	2.86
FS 2525	2.53	0.51	3.04
FS 2900	2.56	0.20	2.76

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APPENDIX A

HONEYCOMB SANDWICH FUSELAGE ASSESSMENT

INTRODUCTION

Preliminary structural analysis was performed to determine the applicability of a honeycomb sandwich shell to a near-term Mach 2.7 supersonic cruise aircraft fuselage design. This investigation included sizing of the honeycomb shell at discrete regions for the design bending moments and shears, and also for the discontinuity stresses caused by the pressure and temperature differential between shell and frame at the operational condition. The resulting weight trends of the sandwich shell were identified and compared to the conventional skin-stringer construction.

POINT DESIGN REGIONS

For the structural-material investigation of the honeycomb sandwich fuselage design, selective regions of the airplane were chosen for analysis and definition of the load-temperature environment. Four point design regions were selected and are shown in Figure A-1 superimposed on the airplane configuration. Those regions selected were located at fuselage stations 750, 2000, 2500 and 3000 and were considered as typical of the critical design regions on the fuselage and, in general, classified as follows:

- Fuselage Forebody (FS 750) - Generally characterized as fatigue-designed structure with low load intensities due to fuselage bending.
- Fuselage Centerbody (FS 2000 and 2500) - Wing/fuselage regions subjected to maximum body bending and wing spanwise loads.
- Fuselage Aftbody (FS 3000) - High body bending and torsion loads with regions subjected to a high acoustic environment.

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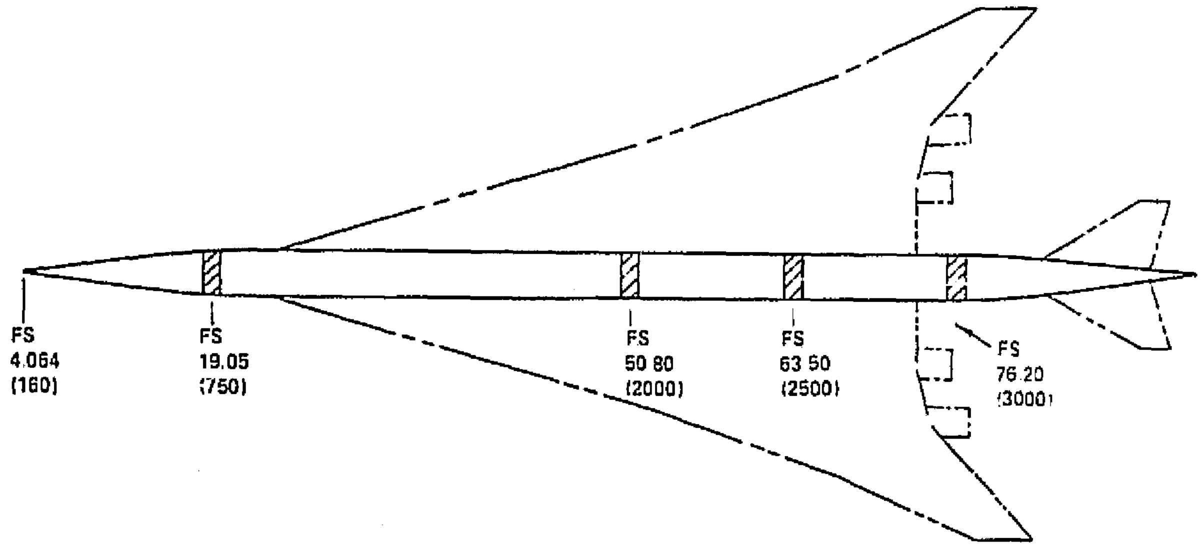


Figure A-1. Definition of Fuselage Point Design Regions

POINT DESIGN ENVIRONMENT

The load-temperature environment was defined at each of the aforementioned point design regions to provide the foundation for the structural analysis and included:

- The load intensities due to the applied fuselage shear and bending moments.
- The normal loads acting on the shell due to internal pressure.
- The average component temperatures and gradients associated with the sandwich design.

The internal loads due to the fuselage shear and bending moments were calculated using the external loads reported in Reference A-1. These external loads are presented in Figures A-2 and A-3, with the following maximum point design values for FS 2000, FS 2500 and FS 3000.

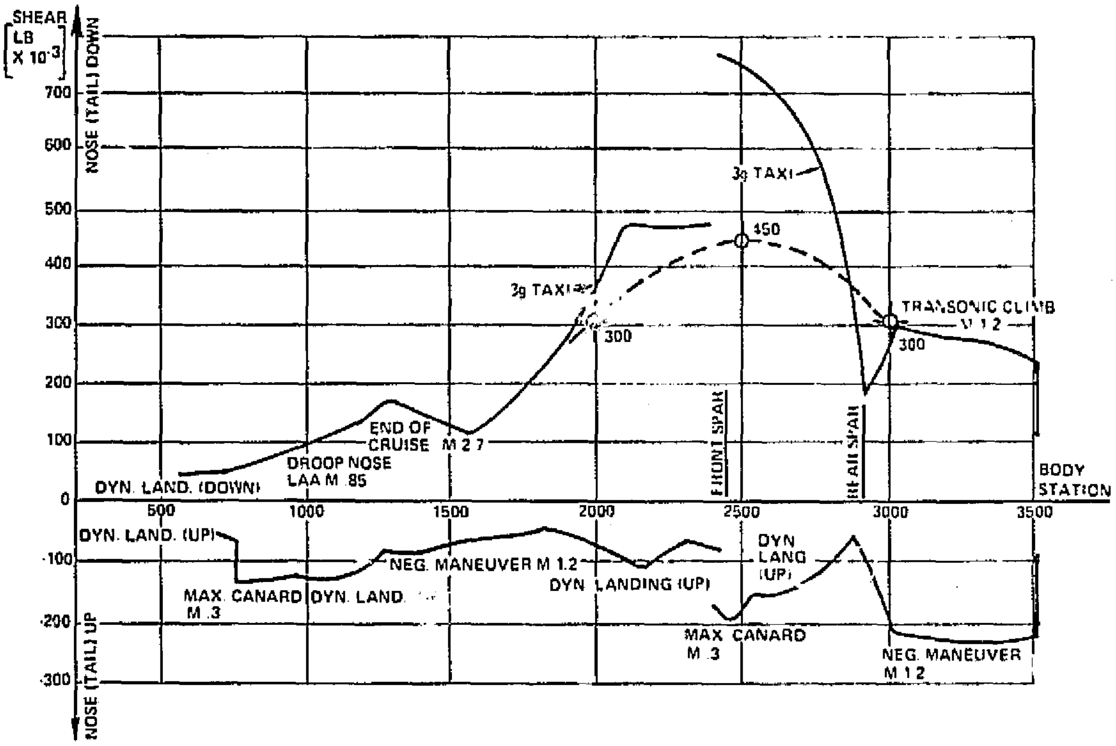


Figure A-2. Fuselage Shear Diagram

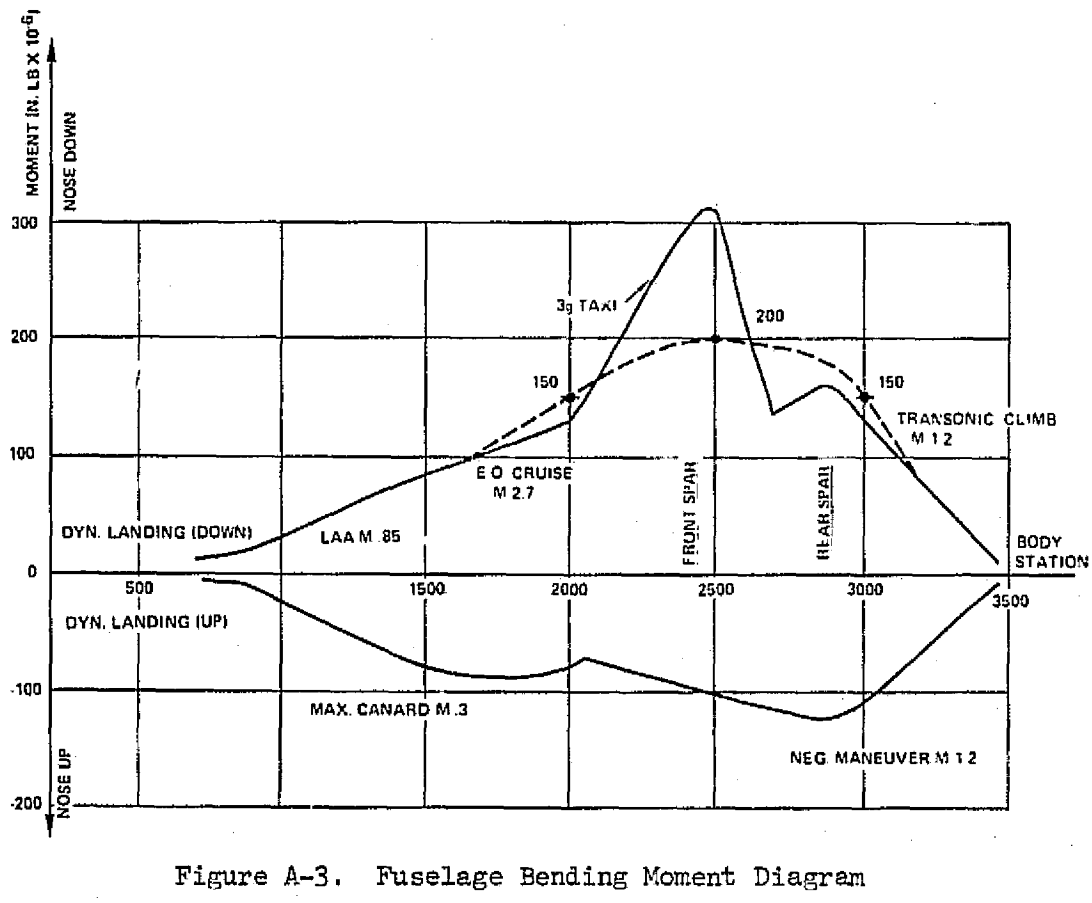


Figure A-3. Fuselage Bending Moment Diagram

Fuselage Station	Ultimate Bending Moment (in-lbs)	Ultimate Shear (lb)
2000	150×10^6	300×10^3
2500	200×10^6	450×10^3
3000	150×10^6	300×10^3

The corresponding internal loads were defined using the above applied loads and theoretical bending (MC/I) and shear (VQ/I) distribution.

Pressurized cabin loads criteria for design differential pressures comply with FAR 25.365 and were taken from Reference A-2. Design pressures are based on providing a 6000 ft. cabin altitude at a flight altitude of 70,000 feet. These conditions produce a nominal cabin pressure of 11.8 psia, which combined with the ambient pressure at 70,000 ft. altitude of 0.6 psia results in a nominal differential pressure of 11.2 psi.

Maximum design differential pressure includes a tolerance which accounts for variations in static reference, a regulator valve tolerance, and relief valve tolerances as illustrated in Figure A-4.

An envelope of differential pressure values used to determine loads on the pressurized cabin is shown on Figure A-5. The limits for structural design range from -0.4 psi to 11.7 psi, with intermediate values between sea level and 38,000 feet. The variation is established by considering a cabin pressure equal to sea level pressure as a limiting value.

A differential pressure varying from -0.4 psi to the appropriate maximum differential pressure for a particular altitude, consistent with the design envelope shown on Figure A-5, is combined with the external air loads and other appropriate structural loads due to maneuvers or gusts. For the operational condition (cruise), a nominal differential pressure of 11.7 psi was used in combination with the thermal environment for evaluating the fatigue strength. The maximum fuselage shell membrane forces (ultimate) due to the internal pressurization are shown in Table A-1. This table contains both the meridional and hoop forces for each of the point design regions.

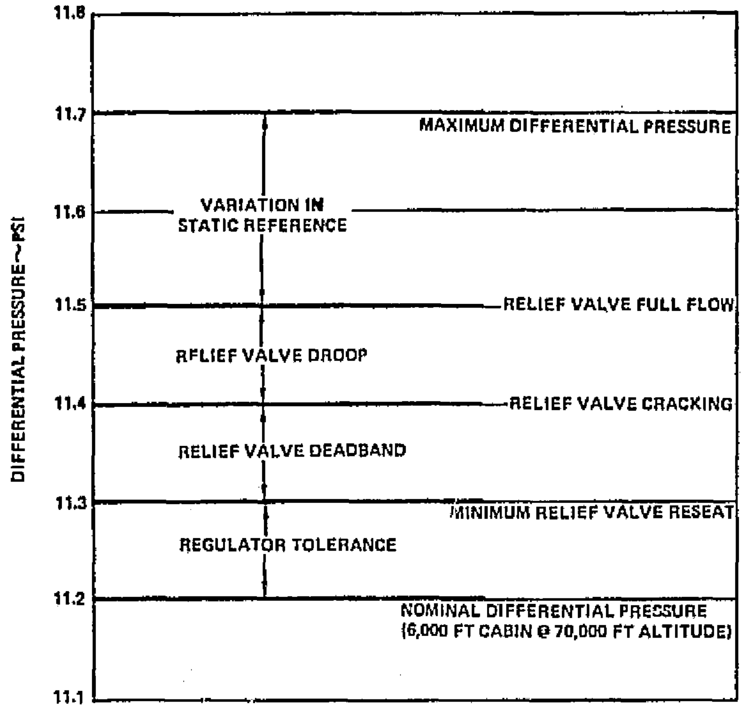


Figure A-4. Tolerances Applied to Nominal Pressure to Established Limit Design Differential Pressure

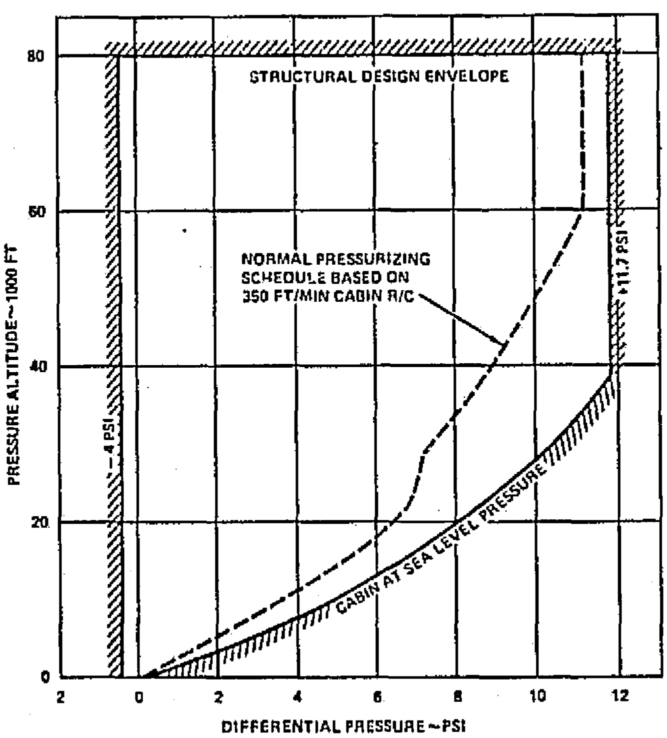


Figure A-5. Cabin Pressure Structural Design Envelope

TABLE A-1. FUSELAGE SHELL MEMBRANE FORCES DUE TO INTERNAL PRESSURIZATION

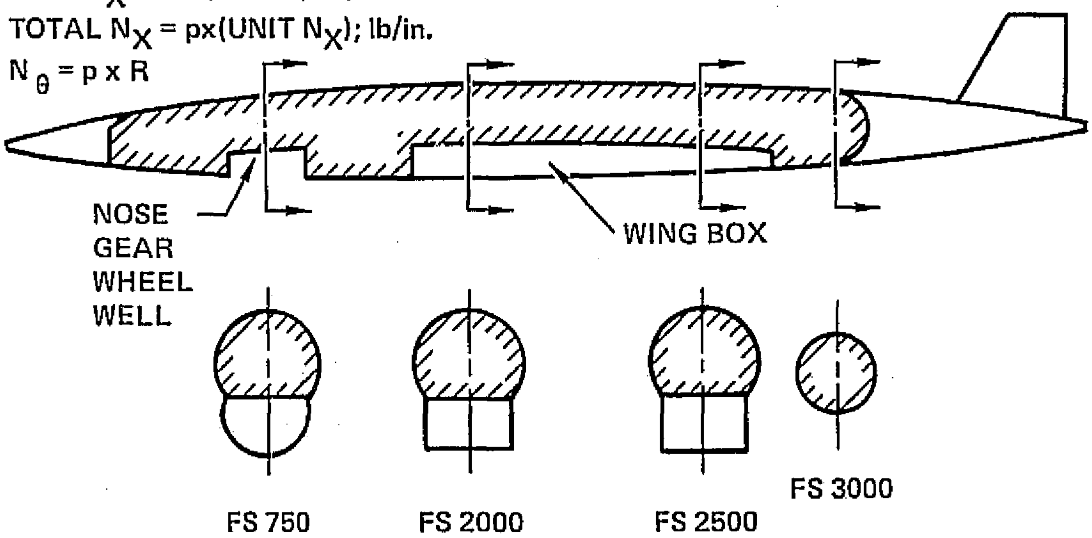
POINT DESIGN REGION	R in.	A in.^{2}	C in.	UNIT N_X (lb/in.)	DESIGN (2) PRESSURE p (PSI)	TOTAL N_X (lb/in.)	HOOP N_θ (lb/in.)
750	72.0	11,761	411	28.6	17.55	502	1264
2000	68.0	10,787	394	27.4	17.55	480	1193
2500	68.0	10,787	394	27.4	17.55	480	1193
3000	61.0	11,690	383	30.5	17.55	535	1070

1. NOMENCLATURE

   R = SHELL RADIUS, in.
   A = ENCLOSED PRESSURIZED AREA, in.^{2}
   C = SHELL CIRCUMFERENCE, in.
   UNIT N_X = A/C, lb/in. per psi
   TOTAL N_X = px(UNIT N_X); lb/in.
   $N_\theta = p \times R$

2. ULTIMATE DESIGN PRESSURE FOR START-OF-CRUISE FLIGHT CONDITION

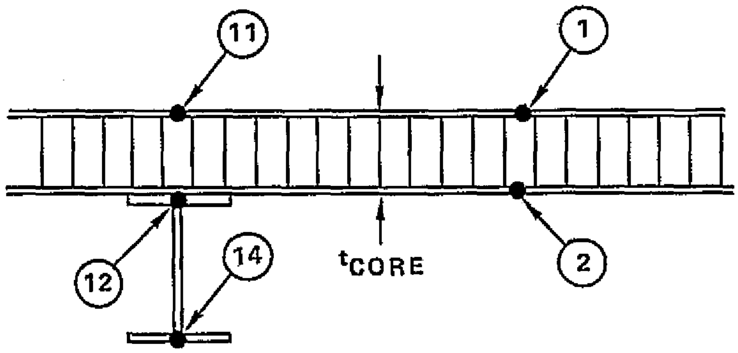
3. PRESSURIZED REGION



Temperature variations through the sandwich shell and frames were calculated using the mission profiles defined in Reference A-2. These data are shown in Table A-2 for the maximum heating climb and maximum cooling descent condition for various core thicknesses.

A summary of the point design environment which includes the inplane loads, normal pressure, and temperatures is presented in Table A-3 for the start-of-cruise condition.

TABLE A-2. FUSELAGE TEMPERATURES, HONEYCOMB SANDWICH DESIGN



0.5-in. CORE: 0.5 X 0.005 X 3/16 HEX, t_f = 0.04-in.

1.0-in. CORE: 1.0 X 0.0015 X 3/16 HEX, t_f = 0.04-in.

1.5-in. CORE: 1.5 X 0.002 X 3/16 HEX, t_f = 0.03-in.

	MAX. HEATING CLIMB MAX. °F END OF CLIMB			MAX. COOLING DESCENT MAX. °F DURING DESCENT		
LOCATION	t_{core} = 0.5	1.0	1.5	t_{core} = 0.5	1.0	1.5
T_1	425	440	438	-31	-28	-27
T_2	408	341	324	-21	22	33
T_{11}	370	414	407			
T_{12}	336	264	239			
T_{14}	86	80	80			

TABLE A-3. FUSELAGE POINT DESIGN ENVIRONMENT, HONEYCOMB SANDWICH DESIGN

START OF CRUISE; MACH NO. 2.7; n_z=2.5

ITEM	UNITS	FS 750			FS 2000			FS 2500			FS 3000		
		UPPER PANEL	SIDE PANEL	LOWER PANEL	UPPER PANEL	SIDE PANEL	LOWER PANEL	UPPER PANEL	SIDE PANEL	LOWER PANEL	UPPER PANEL	SIDE PANEL	LOWER PANEL
N_x	LB/IN	1580	200	-1580	11630	1230	–	15730	1230	–	11630	1230	-11670
N_{xy}	LB/IN	50	250	50	412	1360	–	629	2025	–	412	1360	415
INTERNAL PRESSURE	PSI	17.55	17.55	17.55	17.55	17.55	–	17.55	17.55	–	17.55	17.55	17.55
$T_{AVG}^{(1)}$	°F	416	416	416	390	390	–	390	390	–	390	390	390
$\Delta T^{(2)}$	°F	20	20	20	100	100	–	100	100	–	100	100	100

NOTES:

1. AVERAGE FACE SHEET TEMPERATURE AT MIDBAY
2. TEMPERATURE DIFFERENCE BETWEEN FACE SHEETS AT MIDBAY

DESIGN ALLOWABLES

The tension allowables established to meet the fatigue and fail-safe requirements of Reference A-2 were used for this structural investigation. These requirements were achieved by limiting the gross-area tension stresses for both the ultimate and operational design conditions. For fuselage bending material, the ultimate design gross area tension stress was limited to 90,000 psi, whereas, for the operational condition, the gross area tension stresses were limited to 25,000 psi for the fuselage shell and 35,000 psi for the substructure.

The tension and shear stresses were combined using the principal stress equation and compared to the appropriate gross area tension allowable. The principal stress equation is

$$f_n = \frac{f_x + f_\theta}{2} \pm \left[\left(\frac{f_x - f_\theta}{2}\right)^2 + f_{xy}{}^2\right]^{1/2}$$

where the biaxial stress state is defined by the tension stresses in the axial (f_x) and hoop direction (f_θ), and the shear stress f_{xy}.

Allowable stresses were calculated for both the bending and shear general instability failure modes. The buckling equations and curves defined in Reference A-3 were used to predict the allowable load of the sandwich shell in bending. For the torsional buckling allowable, Reference A-4 was used to define the allowable shear flow.

The interaction formula used to combine the compression and shear loads was the conservative straight-line equation

$$R_b + R_t = 1$$

where the quantities R_b and R_t are, respectively the bending and torsion load ratios.

FUSELAGE WEIGHT COMPARISONS

Weight comparisons are shown for the honeycomb sandwich fuselage design and the conventional skin/stringer design. Both were sized using a common design criteria.

Honeycomb Sandwich Design - A summary of the shell geometry and panel data for this design is shown in Table A-4 with the panel data reflecting the upper shell requirements. Face sheet thickness ranged from a minimum of 0.020 inches at FS 750 to a maximum of 0.092 inches at FS 2500. Identical face sheet thicknesses of 0.070 are noted for FS 2000 and FS 3000. At the centerbody and aft body regions, core height and cell size were held constant at values of 0.75 inches and 0.25 inches, respectively. At the forebody region, FS 750, a core height of 0.500 inches and a cell size of 0.187 inches were used for the design.

A weight summary of the complete panel at each station are shown in Table A-5 and contains the weight attributed to the core, brazing material, and the basic face

TABLE A-4. FUSELAGE PANEL GEOMETRY, HONEYCOMB SANDWICH DESIGN

POINT DESIGN REGION	FS 750	FS 2000	FS 2500	FS 3000
SHELL GEOMETRY				
RADIUS (in.)	72.0	68.0	68.0	61.0
FRAME SPACING (in.)	40.0	40.0	40.0	40.0
PANEL DATA				
HEIGHT, h (in.)	0.500	0.75	0.75	0.75
FACE SHT. THK., t (in.)	0.020	0.070	0.092	0.070
CELL SIZE (in.)	0.187	0.250	0.250	0.250

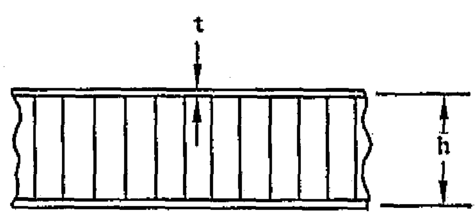


TABLE A-5. SUMMARY OF FUSELAGE PANEL WEIGHTS, HONEYCOMB SANDWICH DESIGN

POINT DESIGN REGION	HEIGHT h (in.)	CELL SIZE S (in.)	EQUIVALENT PANEL THICKNESSES						
			FACE SHEET		BRAZE		CORE		TOTAL
			$\bar{t}$ (in.)	$\bar{t}_F$ (in.)	W_B (lb/ft^2)	$\bar{t}_B$ (in.)	ρ_c (lb/ft^3)	$\bar{t}_c$ (in.)	$\bar{t}$ (in.)
FS 750	0.50	0.187	0.020	0.040	0.22	0.010	8.2	0.014	0.064
FS 2000	0.75	0.250	0.070	0.140	0.20	0.009	8.2	0.020	0.169
FS 2500	0.75	0.250	0.092	0.184	0.20	0.009	8.2	0.020	0.213
FS 3000	0.75	0.250	0.070	0.140	0.20	0.009	8.2	0.020	0.169

$$\bar{t}\,(\text{TOTAL}) = \bar{t}_F + \bar{t}_B + \bar{t}_C$$

where:

$\bar{t}_F = 2t$, in.

W_b = BRAZE WEIGHT, lb/ft^2

$$\bar{t}_B = \frac{W_B}{144\ \rho_F}$$

ρ_c = CORE DENSITY, lb/ft^3

ρ_F = FACE SHEET MATERIAL DENSITY, lb/in^3

$$\bar{t}_C = \frac{\rho_c}{1728}\ \frac{h-t}{\rho_F}$$

h = SANDWICH HEIGHT, in.

t = FACE SHEET THK., in.

sheets. The weight data for the 3003 aluminum braze alloy used for panel fabrication was obtained from empirical data reported in Reference A-5. A core density of 8.2 lb/ft^3 was maintained for all panel designs to preclude any local failure modes (intracell buckling, face wrinkling) or an interaction of these modes with the cylinder-buckling mode. This core density meets the core design criterion, $\delta \geq .03$, recommended in Reference A-3; where, δ is the ratio of core density to face sheet density.

With reference to the panel weights displayed in Table A-5, the formulas used for computing the equivalent thicknesses ($\bar{t}$) and unit weights are shown in the footnotes.

The frame weights for the honeycomb sandwich design are shown in Table A-6 and reflects the equivalent panel thicknesses of the frames for a spacing of 40.0 inches.

Conventional Fuselage Design - The conventional fuselage design was subjected to point design analysis at the same regions using the identical design criteria as used in the sandwich fuselage investigation.

Table A-7 shows the structural concept and corresponding panel dimension for shell design at the most critical circumferential location for each point design region. Equivalent panel thicknesses ranged from 0.056 inches for the zee-stiffened design at FS 750 to 0.184 inches for the hat-stiffened concept at FS 2500. Similarly, the frame equivalent thicknesses are shown in Table A-8 and includes the component (frame and shear-tie) and total thicknesses required at each station.

Table A-9 summarizes the component thicknesses, the total equivalent thickness for each point design region, and the corresponding unit weights for the conventional fuselage design. A maximum weight of 4.75 lb/sq. ft. is indicated for the upper panel at FS 2500; whereas, the similarly located panel at FS 750 weighs 1.54 lb/sq. ft.

Design Comparisons - Table A-10 summarizes the panel and total weight, frame and panel, for both designs. The honeycomb design is heavier than the conventional design at all fuselage stations investigated, with a maximum weight increase of 6-percent noted at FS 2500. This weight increment is attributed directly to the panel weight as the weight of the frame for the conventional design are heavier than those used in the sandwich fuselage design. It is further noted from a

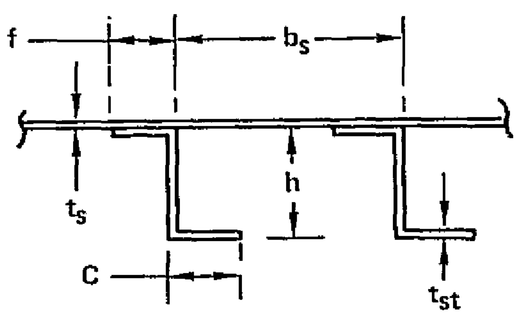
TABLE A-6. FUSELAGE FRAME WEIGHTS, HONEYCOMB SANDWICH DESIGN

POINT DESIGN REGION	CIRCUM. LOCATION	FRAME SPACING L, (in.)	FRAME AREA A, (in.^{2})	EQUIVALENT PANEL THICKNESS t, (in.)
FS 750	UPPER FIBERS	40.0	0.20	0.005
FS 2000	UPPER FIBERS	40.0	0.20	0.005
FS 2500	UPPER FIBERS	40.0	0.20	0.005
FS 3000	UPPER FIBERS	40.0	0.20	0.005

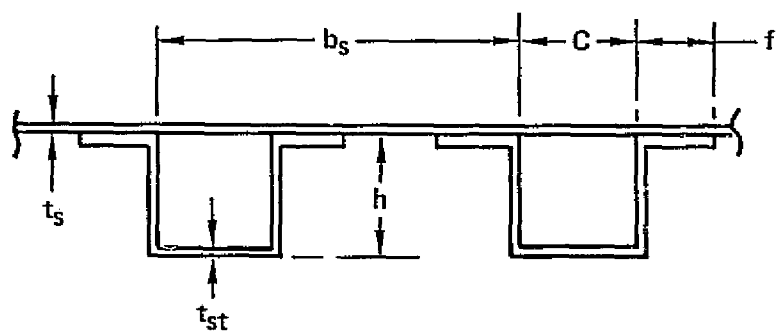
comparison of the basic panel data for each design, Tables A-5 and A-7, that the sum of the face sheets thicknesses for the sandwich design are equal to or less than the corresponding equivalent thicknesses of the skin-stringer design. Hence the parasitic weight of the core and braze alloy overcome any strength/weight advantage of sandwich design, e.g., at FS 2500 equal thickness designs are noted prior to inclusion of the parasitic weight to the sandwich design; whereas, after these items are added to the sandwich design an increase of approximately 16-percent is noted.

TABLE A-7. FUSELAGE PANEL GEOMETRY - CONVENTIONAL DESIGN

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	FUSELAGE PANEL DIMENSION						
			b_s (IN.)	t_s (IN.)	C (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	t (IN.)
FS 750	ZEE-STIFFENED	TOP	4.0	.036	.55	.75	1.00	.036	.056
		SIDE	4.0	.036	.55	.75	1.00	.036	.056
		BOTTOM	4.0	.036	.55	.75	1.00	.036	.056
FS 2000	HAT-STIFFENED	TOP	6.0	.080	1.5	.80	1.25	.070	.145
		SIDE	6.0	.063	1.5	.75	1.25	.040	.099
FS 2500	HAT-STIFFENED	TOP	6.0	.100	1.5	.80	1.25	.090	.184
		SIDE	6.0	.063	1.5	.75	1.25	.050	.109
FS 3000	HAT-STIFFENED	TOP	6.0	.080	1.5	.80	1.25	.070	.145
		SIDE	6.0	.063	1.5	.75	1.25	.040	.099
		BOTTOM	6.0	.090	1.5	.90	1.25	.090	.177



ZEE-STIFFENED CONCEPT



HAT-STIFFENED CONCEPT

TABLE A-8. FUSELAGE FRAME WEIGHTS, CONVENTIONAL DESIGN

POINT DESIGN REGION	CIRCUM. LOCATION	FRAME SPACING (IN.)	EQUIVALENT PANEL THICKNESS ($\bar{t}$), IN.^{2}/IN.			
			FRAME	SHEAR TIE	TOTAL	AVERAGE
FS 750	ALL	20.0	.007	.004	.011	(.011)
FS 2000	UPPER	20.0	.018	.006	.024	(.023)
	SIDE	20.0	.016	.006	.022	
FS 2500	UPPER	20.0	.016	.006	.022	(.022)
	SIDE	20.0	.016	.006	.022	
FS 3000	UPPER	20.0	.018	.006	.024	
	SIDE	20.0	.015	.006	.021	(.023)
	LOWER	20.0	.019	.007	.026	

$\bar{t}$(TOTAL) = $\bar{t}$(FRAME + $\bar{t}$(SHEAR TIE)

FRAME GEOMETRY

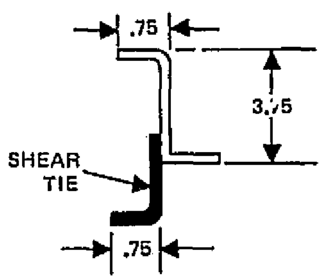


TABLE A-9. FUSELAGE WEIGHT SUMMARY, CONVENTIONAL DESIGN

POINT DESIGN REGION	PANEL CONCEPT	EQUIV. PANEL THICKNESS (IN.^{2}/IN.)			UNIT WEIGHT W (LB/SQ. FT)
		FRAME $\bar{t}$	PANEL $\bar{t}$	TOTAL $\bar{t}$	
FS 750	ZEE-STIFF.	0.011	0.056	[illegible]	1.54
FS 2000	HAT-STIFF.	0.024	0.145	0.169	3.89
FS 2500	HAT-STIFF.	0.022	0.184	0.206	4.75
FS 3000	HAT-STIFF.	0.024	0.145	0.169	3.89

NOTE: THICKNESS AT UPPER CIRCUMFERENTIAL LOCATION SHOWN

$$W = 144 \times \rho \times \bar{t} \, (\mathrm{lb/ft^2})$$

WHERE: $\rho = 0.170$ lb/in.^{3}

TABLE A-10. COMPARISON OF FUSELAGE SHELL WEIGHTS, CONVENTIONAL AND HONEYCOMB SANDWICH DESIGNS

POINT DESIGN REGION	FUSELAGE SHELL WEIGHT, lb. sq. ft.			
	CONVENTIONAL		HONEYCOMB SANDWICH	
	PANEL	TOTAL	PANEL	TOTAL
FS 750	1.29	1.54	1.47	1.59
FS 2000	3.34	3.89	3.89	4.01
FS 2500	4.24	4.75	4.91	5.02
FS 3000	3.34	3.89	3.89	4.01

REFERENCES

A-1 Anon: "Mach 2.7 Fixed Wing SST Model 969-336C (SCAT-15F)" D6A 11666-1, The Boeing Company, 1969

A-2 Anon: "Supersonic Transport Development Program - Phase III Proposal, Volume II-C Airframe Design" LR 19839, Lockheed-California Co., 1966

A-3 Anon: "Buckling of Thin-Walled Circular Cylinders" NASA SP-8007, August 1968

A-4 Becker, H: "Handbook of Structural Stability, Part VI - Strength of Stiffened Curved Plates and Shells" NACA TN 3786, July 1958.

A-5 Elrod, S.D. and Lovell, D.T.: "SST Technology Follow-on Program, Phase I - Development of Aluminum Brazed Titanium Honeycomb Sandwich Structure" FAA-SS-72-03, 1972.

SECTION 13

FATIGUE AND FAIL-SAFE ANALYSIS

BY

S.T. CHIU, G.W. DAVIS and J.C. EKVALL

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A	Cross-sectional Area
A_e	Effective Area of the Reinforcement
A_R	Cross-sectional Area of Reinforcement
A_{st}	Cross-sectional Area of Stiffener
a, b	X and Y Distance between Simply Supported Edges of Panel
F_g, F_s	Allowable Tensile and Shear Stress
F_{tu}	Material Ultimate Tensile Strength
f_x, f_y, f_{xy}	Inplane Stresses Associated with the x-y Plane
f_θ	Hoop Stress
K_c, k_o	Fracture Toughness Allowable $K_c = 1.25\ k_o$
K_Q	Fatigue Quality Index
K_t	Stress Concentration Factor
L	Crack Length
N	Number of Cycles to Failure
n	Curvature Reduction Factor
P	Force
R	Stress Ratio, $R = S_{min}/S_{max}$; Radius
S_{eq}	Equivalent Stress
S_{max}, S_{min}	Maximum Stress; Minimum Stress
S_x, S_y, S_{xy}	Inplane Stresses Associated with the x-y Plane
t	Material Thickness
$\overline{t}$	Equivalent Panel Thickness
$\overline{y}$	Distance from the Inner Surface of the Sheet to the Centroid of the Stiffener

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LIST OF SYMBOLS (Continued)

γ	Reinforcement Efficiency Parameter
ρ	Radius of Gyration
ψ	Shear Correction Factor

SECTION 13

FATIGUE AND FAIL-SAFE ANALYSIS

INTRODUCTION

Analyses were conducted to establish design stress levels for fatigue and fail-safe evaluation of structural design concepts for an arrow-wing supersonic cruise aircraft configuration. The primary structure was evaluated to meet the specific service life of 50,000 flight hours and to support the fail-safe design load of 100 percent limit load. Related design criteria as specified in the Federal Aviation Agency FAR 25, Airworthiness Standards and the supplemental tentative Airworthiness Standard for Supersonic Transports were used as the basis for this evaluation.

A description of design criteria, and the results of the fatigue, crack growth, and fail-safe analyses are presented in the following text.

DESIGN CRITERIA

All commercial aircraft must be designed to meet Federal Aviation Agency FAR 25, Airworthiness Standards: Transport Category Airplanes. For an advanced supersonic transport, additional special provisions, similar to the tentative Airworthiness Standards for the Supersonic Transport, will be specified prior to the design of such an aircraft. These criteria specify that the flight structure whose failure could result in catastrophic failure of the airplane must be evaluated to meet either the fatigue strength requirement, Section 25.571(b), or the fail-safe strength requirement, Section 25.572(c). The wing, fuselage and empennage structure of all commercial aircraft are generally designed to comply with Section 25.571(c) and therefore fatigue substantiation according to Section 25.571(b) is not required. However, the structure is designed and generally fatigue tested to demonstrate to the customers (airlines) that no major fatigue problems will occur during the service life of the aircraft. Therefore, in this study the various design concepts were sized to meet both fatigue and fail-safe strength requirements.

No requirements are currently specified for crack growth. However, crack growth analyses were conducted to show that small cracks that are likely to be missed on a given inspection will not grow to catastrophic failure before the next inspection period which is of the order of 8,000-12,000 flight hours.

Fatigue Design Criteria

The basic fatigue design criteria for this program is to provide a structure that will be good for a service life of 50,000 flight hours. Appropriate multiplying factors are applied to the design life for use in establishing allowable design stresses as discussed in Section 4. For structure subjected to spectra loading, the allowables are selected using a factor of 2 times the service life of 50,000 hours. For areas of the fuselage structure subjected to constant amplitude loading the allowable stresses are selected for 200,000 flight hours (50,000 x 4). A larger factor is applied to constant amplitude loading because the scatter in fatigue test data is larger for this type of loading.

Fail-Safe Design Criteria

Fail-safe designs are employed for the wing and fuselage structures which must be capable of supporting the fail-safe design load of 100 percent Limit Load, as defined in the tentative Airworthiness Standards for SST, for the damage cases summarized below.

<u>General</u>

- Any single member completely severed. For fail-safe purposes, a single member is any redundant structural member, or that part of any member, of several elements where the remaining part can be shown to have a high probability of remaining intact in the event of the assumed failure. It must be demonstrated that the damage to the assumed severed part can be discoverable by normal inspection methods.
- Extensive structure severed between the boundaries of effective crack barriers. A mechanical splice (not welded) or major structural members (frame, fail-safe straps or stringers) which are mechanically

fastened to the skin are considered to be effective crack barriers. Tests must be conducted to demonstrate that bonded or brazed reinforcements are effective crack barriers.

- In extensively stiffened skin structure, a major structural member, attached directly and continuously to the skin, fractured, together with the skin between adjacent crack barriers.
- For skin surfaces with no effective crack barriers (splices, stringers, fail-safe straps, etc.), the structure must be capable of supporting the fail-safe load with a 20 inch skin crack using "B Basis" fracture toughness allowables for appropriate temperatures, grain direction and material thickness.
- Welded and laminated skin structure must be considered monolithic for the purpose of fail-safe design. Welded joints cannot be considered as crack stoppers.
- All fail-safe joints and skin splices shall be designed to have sufficient shear lag to distribute loads from the failed section. This can be achieved by:
  - (a) Designing the joint to be bearing critical.
  - (b) Providing sufficient margin in fastener shear strength so that progressive failure of the fasteners will not occur prior to skin and reinforcement failure.

Wing Structure - The wing structure is designed to meet 100 percent Limit Load requirements in the presence of the damage conditions specified below:

- Completely failed shear web of a rib, a spar or a bulkhead.
- Any single member of a truss.
- Failed rib cap or any other element of the rib.
- Failed spar cap.
- For stiffened skin construction the following damage conditions shall apply
  - (a) One to three failed stringers together with a skin crack between adjacent intact stringers. The number of failed stringers depends

on the stringer spacing. Skin crack sizes should be of the order of 6-10 inches long.

(b) A spar cap or other spanwise reinforcing member, attached directly and continuously to the skin, completely severed with a chordwise skin crack between adjacent stringers.

(c) A chordwise reinforcing member, attached directly and continuously to the skin, failed along with a 20 inch spanwise skin crack using "B-Basis" allowables for the skin material. Members with flexible attachments to the skin (through clip) need not be considered broken with a skin crack.

- For sandwich-type construction (monocoque) the following shall apply:

  (a) A major reinforcing element (fail-safe strap, stringer, etc.) attached directly and continuously to a sandwich skin surface, failed, together with skin cracks in both skins between intact adjacent barriers.

Fuselage Structures - Fail-safe requirements for the fuselage structure are met using normal relief valve pressure setting plus external air loads and fail-safe limit design load for the following damage conditions:

- Any of the applicable General or Wing Structure conditions described above.
- For stiffened skin construction:

  (a) A typical frame broken together with a longitudinal skin crack between adjacent fail-safe straps.

  (b) A fail-safe strap broken together with a longitudinal skin crack between adjacent intact frames.

  (c) A single stringer failed along with a circumferential skin crack between intact stringers.

- A main frame completely severed.

FATIGUE ANALYSIS

Wing Structure

Preliminary fatigue lives were calculated for spanwise bending loads acting on the wing structure. These lives are plotted on Figure 13-1 as a function of ultimate design gross area stress and fatigue quality index. The curves on this figure are developed using the concept of linearly cumulative damage with an average flight of two and a half hours; a once-per-flight peak-to-peak ground-air-ground cycle; the climb, cruise, descent and taxi loading of Spectra "C" (see Section 4); and the standardized constant-life diagrams for axial loading of Ti 6Al-4V sheet and plate shown on Figures 13-2 through 13-5.

The calculated lives shown on Figure 13-1 led to the selection of an ultimate design gross area stress level of 90 ksi and a design fatigue quality index of K = 5. This selection is somewhat more conservative than specified in Section 4, since the calculated life equals or exceeds 45,000 flights or 112,500 flight hours rather than 100,000 flight hours as specified in Section 4. Curves showing ultimate design stress versus fatigue quality are shown on Figure 13-6 and the fatigue quality and design allowable stresses for the various design concepts are summarized on Table 13-1.

The ultimate design stresses shown on Figure 13-6 are applicable for general wing structure subjected to spanwise bending and fuselage bending material.

Fuselage Structure

Figure 13-7 presents the relationship between fuselage circumferential design stress and fatigue quality for 50,000 hours of service based on an average of two and a half hours, one pressure cycle per flight and a life reduction factor or 4 which is applicable for constant amplitude loading. In Figure 13-7, the maximum design tension stress corresponds to twice the value of the variable stress for a once-per-flight peak ground-air-ground cycle, $f_{vary(OPFP\ GAG)}$, for R=0 and N=100,000 cycles to failure on S-N diagrams for Ti 6Al-4V (mill annealed) sheet and plate with F_{tu} = 135-155 ksi.

The operational design gross-area tension stresses shown on Figure 13-7 commensurate with a fatigue quality index (K_Q) of 5 were used for the fuselage analysis, i.e., 25 ksi for fuselage skin circumferential stresses and 35 ksi for the substructure (frame) stresses.

C.4

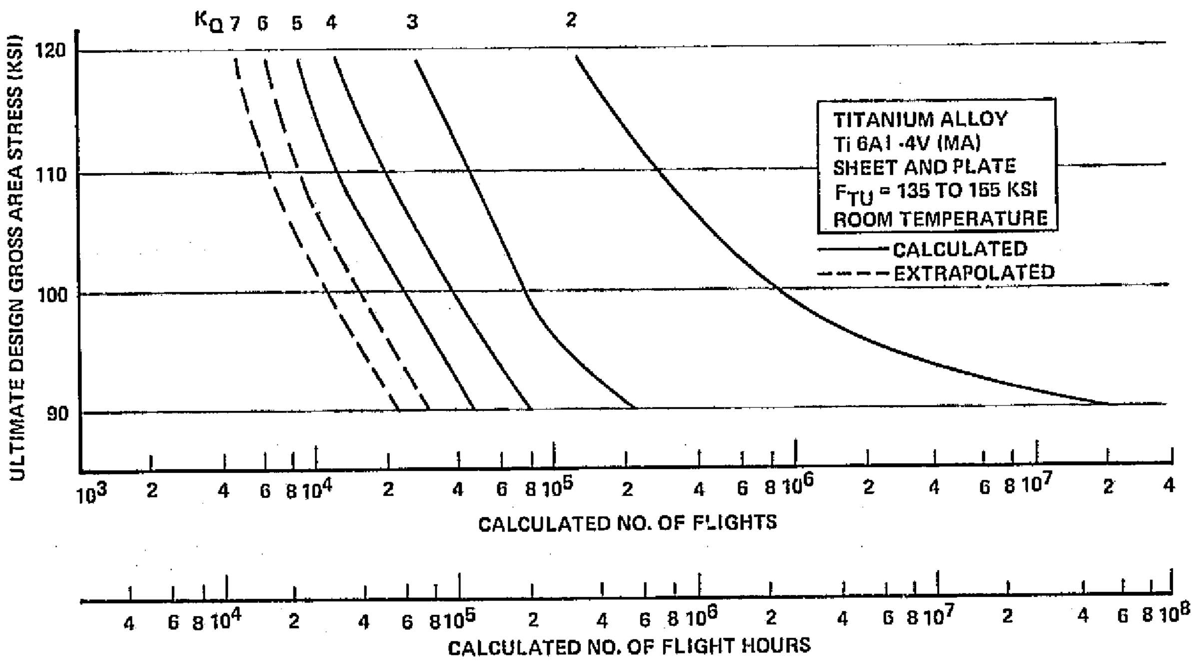


Figure 13-1. Variation in Wing Ultimate Design Gross Area Stress with Life

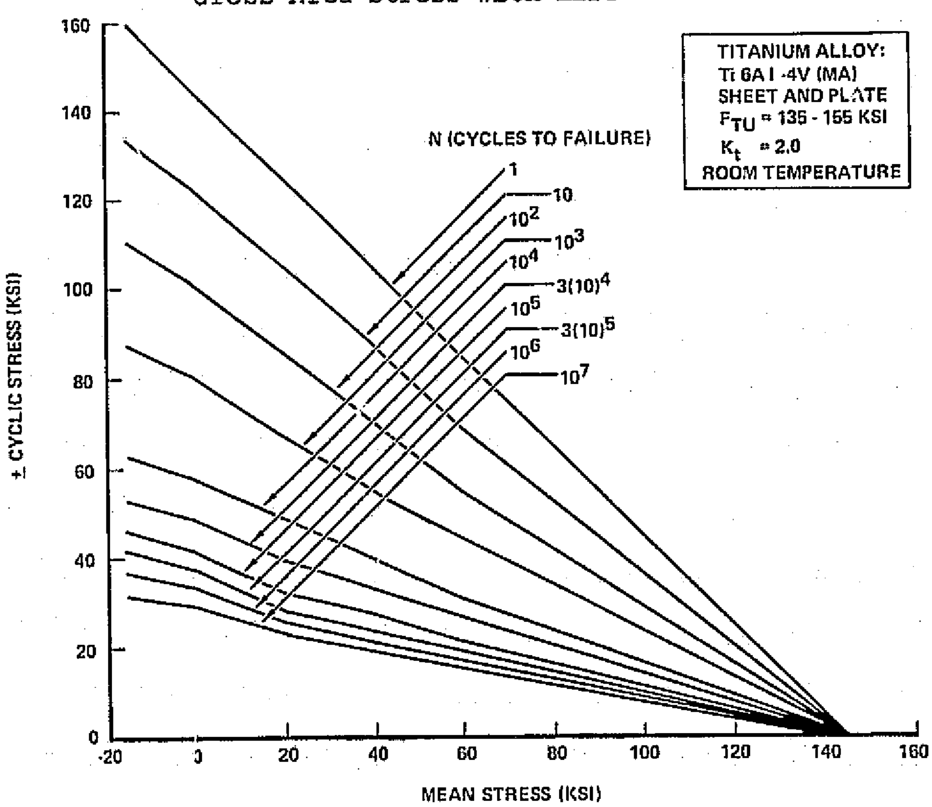


Figure 13-2. Constant Life Diagram - Titanium Alloy (K_t=2)

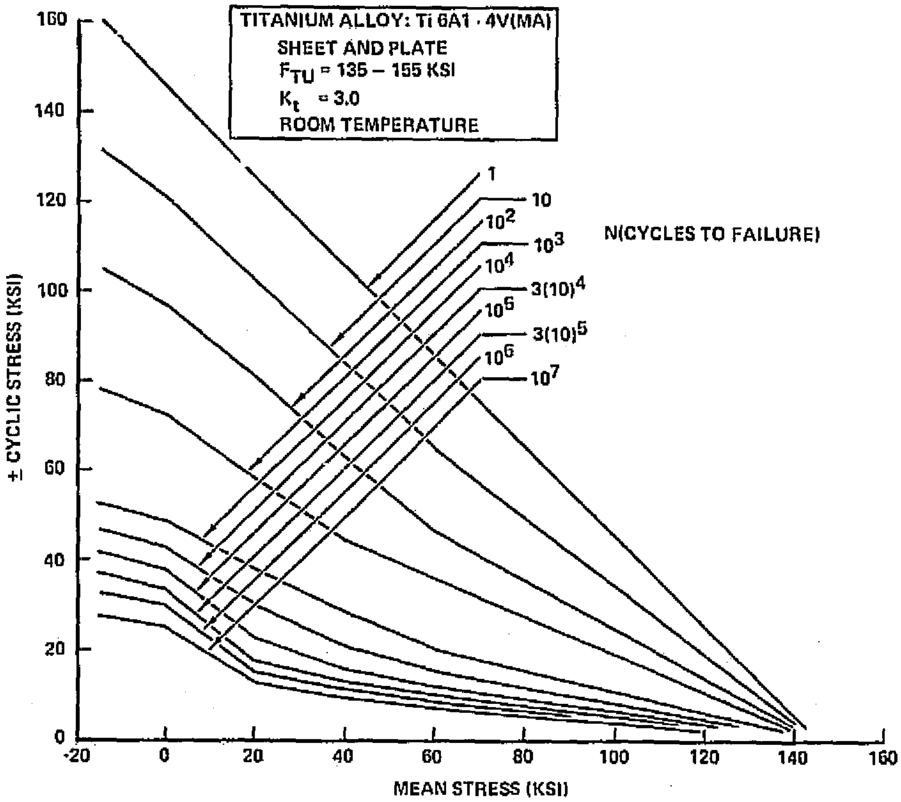


Figure 13-3. Constant Life Diagram - Titanium Alloy (K_t = 3)

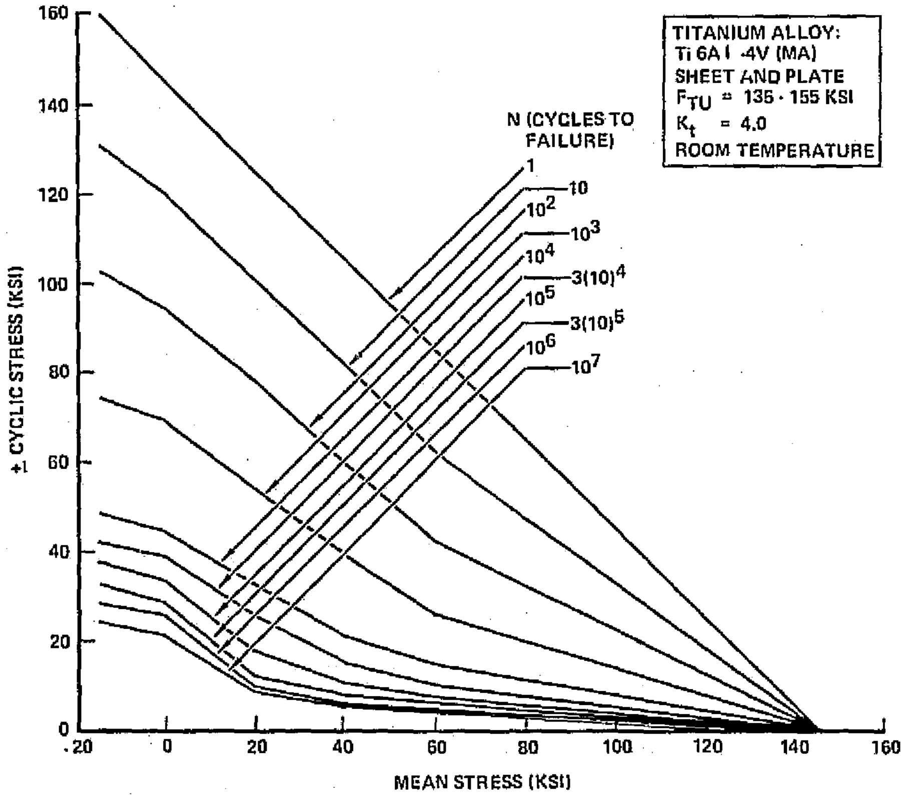


Figure 13-4. Constant Life Diagram - Titanium Alloy (K_t = 4)

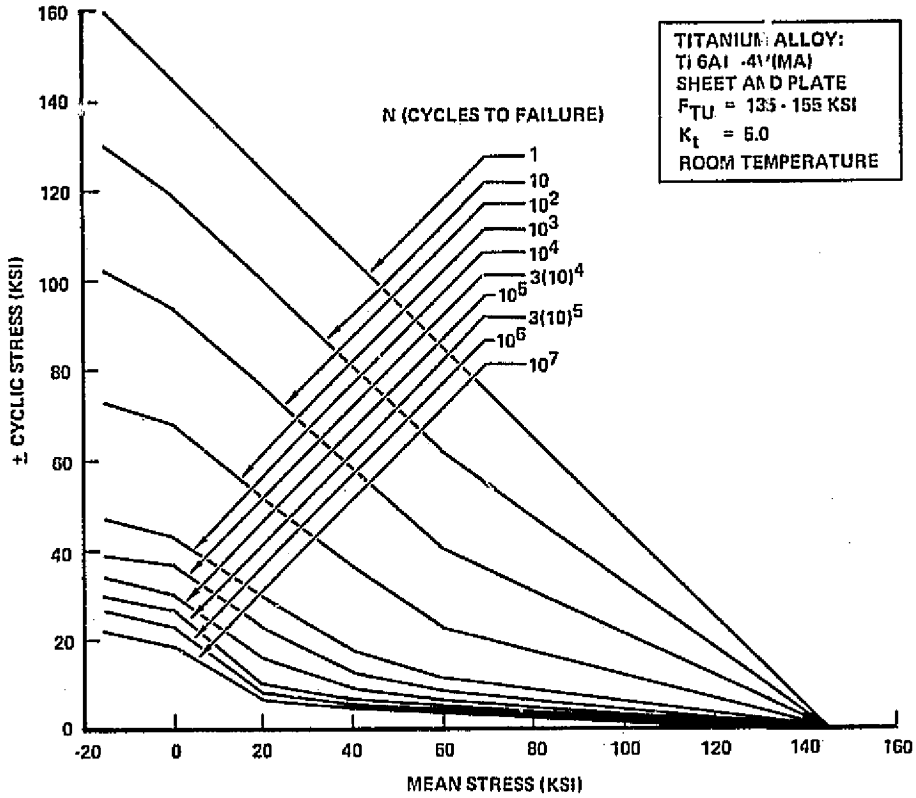


Figure 13-5. Constant Life Diagram - Titanium Alloy (K_t = 5)

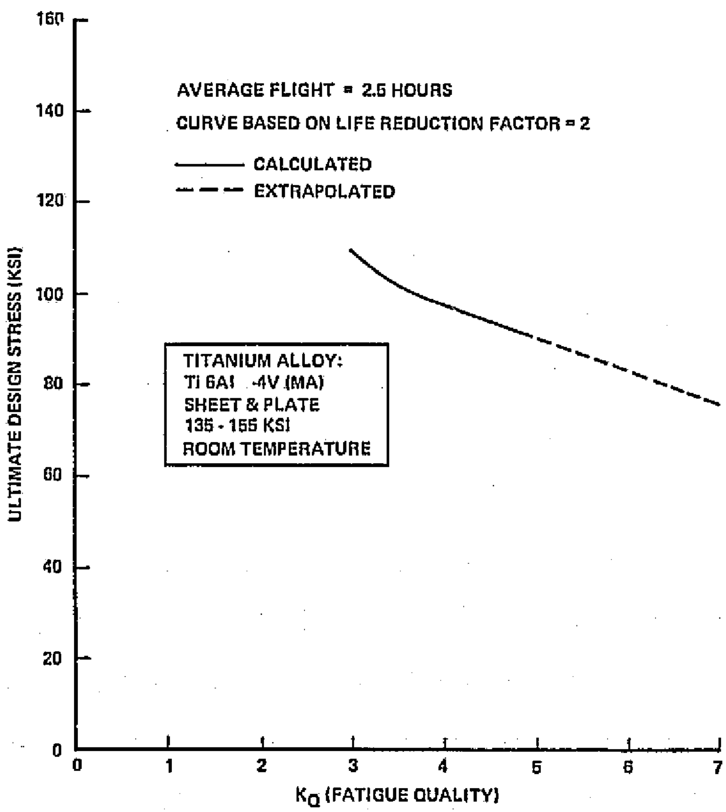


Figure 13-6. Variation in Wing Ultimate Design Stress with Fatigue Quality

TABLE 13-1. SUMMARY OF FATIGUE ALLOWABLES FOR WING STRUCTURE

	K_Q	ULTIMATE DESIGN STRESS, PSI
CHORDWISE STIFFENED		
• PANELS WELD BOND	4	97,000
• JOINT – BOND FEATHER EDGE OUTER SHT.	7 (LOCALLY)	75,000

	K_Q	ULTIMATE DESIGN STRESS, PSI
• SUBSTRUCTURE	5	90,000
• SPARS	5	90,000
SPANWISE STIFFENED		
• CLEAN AREAS	4	97,000
MONOCOQUE		
• CLEAN PANEL AREA	4	97,000
• LOCAL MECH JOINT	5	90,000
• ALL WELDED	4	97,000

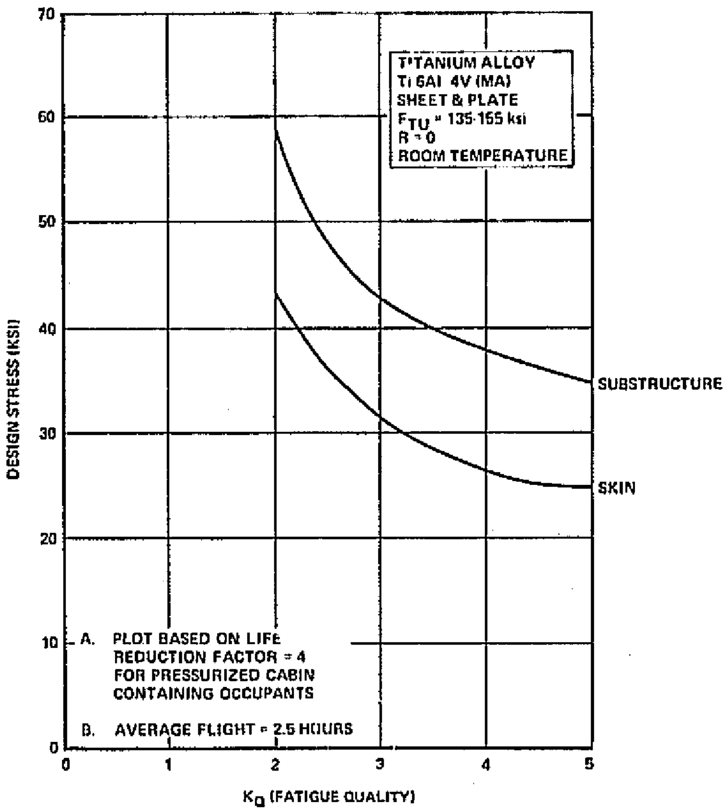


Figure 13-7. Variation in Fuselage Circumferential Design Stress with Fatigue Quality

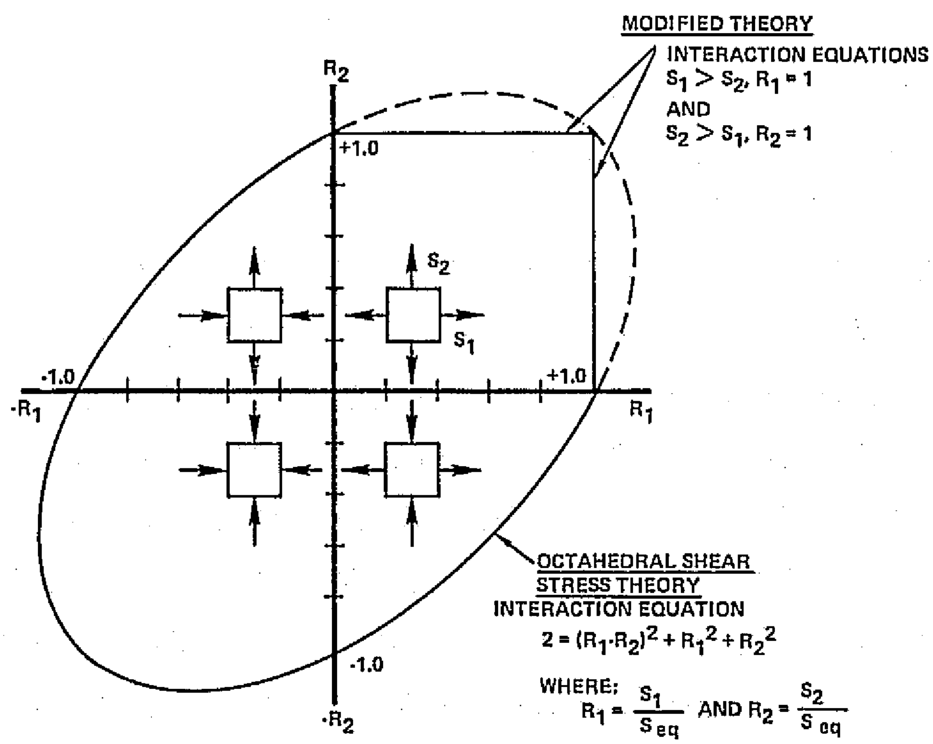


Figure 13-8. Interaction Curve for Combined Loading

The allowable hoop stress for the skin is approximately 28-percent lower than the frame allowable since the skin is subjected to biaxial stresses due to pressure and thermal loads, whereas, the frames are primarily uniaxially loaded.

For fuselage bending material, the ultimate design gross area stress is limited to 90,000 ksi for a fatigue quality index of 5. This value is shown on Figure 13-1 and is applicable to both wing lifting surfaces and fuselage bending material.

Interaction Equations

For shell structure and other areas of the airframe subjected to biaxial and/or shear loads, the Octahedral Shear Stress Theory is used to calculate the applied stress level for fatigue analysis. In this theory, the equivalent axial stress (S_{eq}) for a biaxial stress field in terms of the x and y stress components is as follows:

$$S_{eq} = \left(S_x^2 + S_y^2 - S_x S_y + 3\, S_{xy}^2\right)^{1/2}$$

where S_x, S_y are the direct stresses in the x and y directions, respectively and S_{xy} is the shear stress in the x-y plane.

Or the equivalent stress may be stated in terms of the principal biaxial stresses

$$S_{eq} = \left(S_1^2 + S_2^2 - S_1 S_2\right)^{1/2}$$

where the principal stresses (S_1 and S_2) are given by:

$$S_{1,2} = \frac{S_x + S_y}{2} \pm \left[\left(\frac{S_x - S_y}{2}\right)^2 + S_{xy}^2\right]^{1/2}$$

For this analysis, the ultimate tensile stress calculated using the Octahedral Stress Stress Theory (S_{eq}) was not allowed to exceed the maximum principal stress, max (S_1, S_2).

$$Seq. \leq \max (S_1, S_2)$$

These equations have been visually displayed in Figure 13-8 in terms of interaction equations. Quadrant 1 displays the stress state when both principal stresses are tension and the equivalent stress is constrained so as not to exceed the maximum principal stress. For pure plane shear, mid-point of quadrant 2 and 3, the Octabedral Shear Stress Theory predicts an equivalent stress that is equal to 58-percent of the stress level for an unidirectioned axial load.

CRACK GROWTH ANALYSIS

A preliminary analysis was performed to investigate the fatigue crack growth rate behavior of Ti 6Al-4V, mill-annealed plate when subjected to the Spectra C loading history (reference Section 4). The analysis results reported indicate the effects of design stress and environment on crack growth.

For crack growth prediction it must be possible to obtain an expression for the stress intensity K which characterizes the severity of the local stresses and deformations at the crack tip. For the present analysis, the configuration analyzed was a standard through-thickness crack of length 2a in the center of a wide flat panel, subjected to a uniform gross area tension stress S. The stress intensity for this configuration is given by:

$$K = S\sqrt{\pi a} \tag{1}$$

For fatigue crack growth analysis, an effective cyclic stress can be defined (Reference 1) to characterize the tendency of the fatigue cycle to cause crack growth. For a fatigue cycle with a maximum tensile stress (S_{max}) and a minimum-to-maximum stress ratio (R), the effective cyclic stress is given by (Reference 2)

$$S_{cyc} = [1 - \mathrm{MAX}\ (R, R_c)]^m\ S_{max} \tag{2}$$

where m and R_c are empirical constants and MAX (R,R_c) takes the value of the larger of its arguments. For Ti 6Al-4V, m = 0.75 and $R_c = -1$.

The crack growth resistance of the material is usually described by a functional relationship between the crack growth per cycle and the effective cyclic stress intensity from a constant amplitude fatigue test:

$$da/_{dN} = f\ (K_{cyc}) \tag{3}$$

The effective cyclic stress intensity K_{cyc} can be calculated by using S_{cyc} from Equation (2) in place of S in the stress intensity expression such as Equation (1). For a particular material, product form and thickness, grain direction, chemical environment, cyclic frequency and temperature the function f in (3) is unique.

Table 13-2 lists points on a plot of $da/_{dN}$ vs. K_{cyc} for mill annealed Ti 6Al-4V thin sheet in laboratory air and 3.5 percent NaCl solution for cyclic frequencies of approximately 10 Hz. For any intermediate value of K_{cyc} the corresponding value of da/dN can be found by linear interpolation on log (da/dN) vs. log (K_{cyc}).

A variable-amplitude sequence of cycles occurs for aircraft in service. If the interaction between different loading cycles in the sequence is neglected the increment of growth caused by the jth loading cycle is

$$(\Delta a)_j = f\left(K_{cyc}\right)_j \tag{4}$$

where f is the constant amplitude crack growth rate function, Equation (3), exemplified by Table 13-2.

For many loading spectra the loading cycles do interact. The major effect that has been observed is retardation of crack growth following a high tensile loading. Various investigations (References 3 and 4) have proposed simple retardation models to include this effect in the crack growth calculation.

TABLE 13-2. CRACK GROWTH RATE, Ti 6Al-4V SHEET

K_{cyc} (ksi-$\sqrt{\text{inch}}$)	CRACK GROWTH RATE, da/dN (Microinch/Cycle)	
	NaCl Environment	Air Environment
30	85	32.5
40	100	54
56	130	130
75	370	370
95	3,500	3,500
115	34,000	34,000
140	600,000	600,000

Figure 13-9 shows a scaled sketch of the once-per-flight maximum cyclic stresses during taxi, climb, cruise, and descent for Spectra "C" of Reference 5, for a 1-g stress of 25 ksi. Note that the peak-to-peak GAG cycle is several times greater in amplitude than any other cyclic loading that occurs within a flight. Therefore, for expediency in this preliminary analysis all cycles were neglected except the once per flight peak-to-peak GAG cycle. An approximate analysis has shown that, of the cycles given in Figure 13-9, the GAG cycle causes about 95-percent of the calculated crack growth damage. It is unknown how much additional crack growth might be caused by the very low amplitude gust cycles which would occur in large numbers in service, but which were deleted from Figure 13-9 for analysis of fatigue crack initiation.

For an initial crack length of 2a = 0.25 inch the calculated crack growth lives based on GAG cycles only are shown in Table 13-3. These calculated lives do not reflect real-time, real-temperature effects on crack growth, except that the loading spectrum contains thermally-induced stresses which contribute to the once-per-flight peak stress.

FAIL-SAFE ANALYSIS

General

The objective of the damage tolerance analysis was to ensure that structures in the presence of an assumed damage condition are capable of supporting the damage-tolerance design load of 100-percent limit load.

The analysis method used is presented in Reference 6. Figure 13-10 outlines the method and available data used in determining the residual strength of damaged reinforced structure. Figure 13-11 shows the idealized reinforced panels with two-bay and multi-bay damages. Essentially the method provides reinforcement efficiency for stiffened flat panels based on the given reinforcement spacing and area. A CPS (conversational Programming System) program was used to facilitate the computational procedure. In addition, the margin of safety is also calculated by the program based on the applied design limit load. A resultant positive margin of safety indicates that the structure analyzed is capable of withstanding the imposed damage. On the other hand, a negative margin of safety results in a corresponding weight penalty. This is due to the additional reinforcing straps required for the

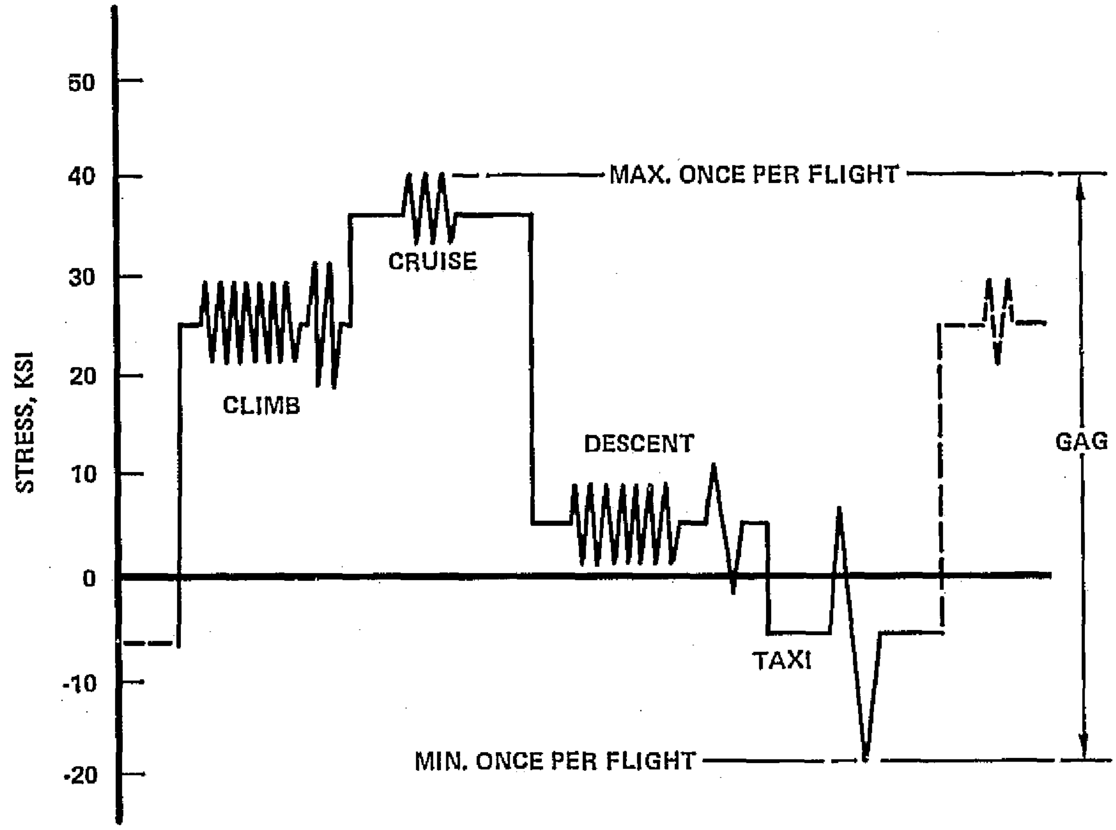


Figure 13-9. Test Flight Pattern Illustrating Peak-to-Peak GAG Cycle for 1-g Design Stress Equal to 25 ksi and Service Flight Time of 2.5 Hours

TABLE 13-3. RESULTS OF PRELIMINARY CRACK GROWTH ANALYSIS

1-g STRESS (Ksi)	ENVIRONMENT	CRACK GROWTH LIFE, $\ell_0 = .25$ FLIGHTS	HOURS
25	Lab. Air	4560	11400
25	3.5% NaCl	3550	8875
30	Lab. Air	2380	5950
30	3.5% NaCl	2080	5200

NOTE: INSPECTION INTERVAL 8000 - 12000 FLIGHT HOURS

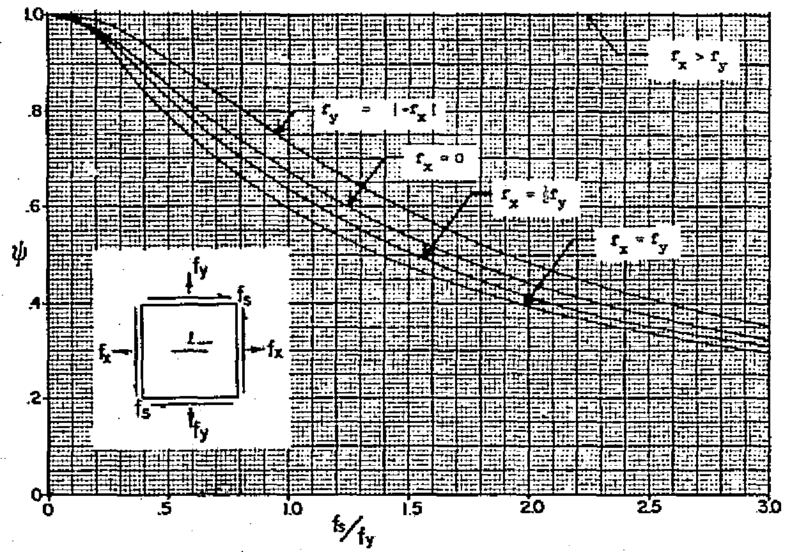


FIGURE 13-10a

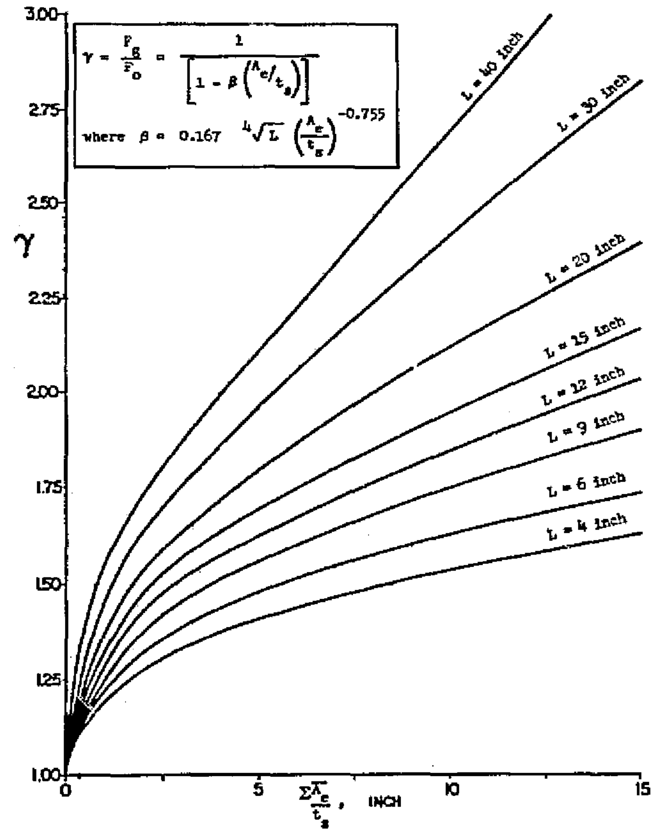


FIGURE 13-10b

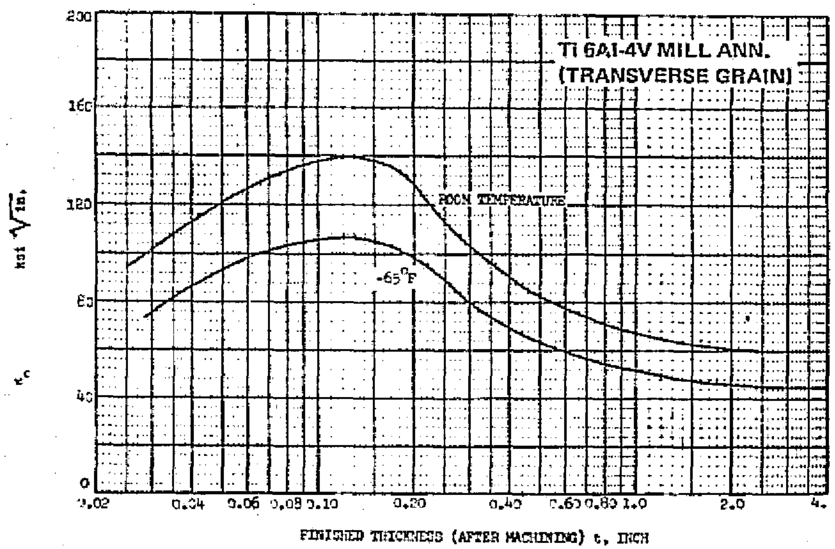


FIGURE 13-10c

ALLOWABLES STRESS (TENSION): $F_g = \frac{\psi\gamma \; (nk_o)}{\sqrt{L}}$

WHERE:

γ = REINFORCEMENT EFFICIENCY PARAMETER, DIMENSION LESS; FIGURE 13-10a

ψ = SHEAR CORRECTION FACTOR, DIMENSIONLESS; FIGURE 13-10b

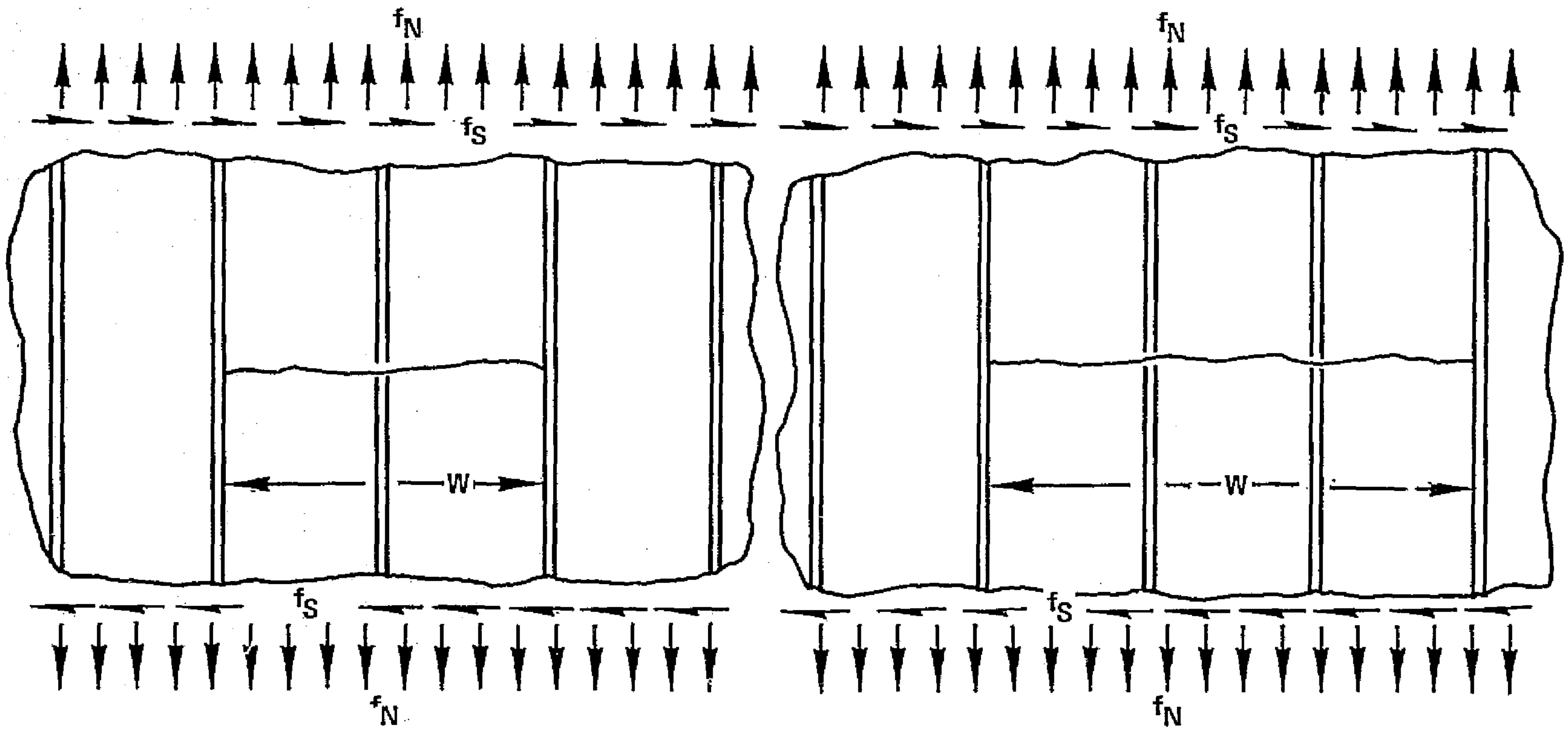
n = CURVATURE REDUCTION FACTOR, DIMENSIONLESS; n = 1 FOR ALL CASES EXCEPT FOR LONGITUDINAL CRACKS IN SHELL STRUCTURE WHERE n = ½

k_o = STRESS INTENSITY FACTOR FOR CONDITIONS OF PLANE STRESS USED FOR THROUGH-THE-THICKNESS CRACK CONFIGURATION; $k_o = K_c / \sqrt{\pi/2}$; FIGURE 13-10c

K_c = PLANE STRESS FRACTURE TOUGHNESS (ASTM NOTATION)

L = TOTAL CRACK LENGTH, L = 2a

Figure 13-10. Method of Determining the Strength of Damaged Structures



(A) TWO-BAY CRACK

(B) MULTIPLE-BAY CRACK

Figure 13-11. Panel Damage Configurations

particular structure to achieve an acceptable margin of safety. Weight penalties are calculated in terms of equivalent surface panel weight. A significant weight penalty will affect the local stress distribution and a redesign might be necessary.

The minimum damaged condition assumed for reinforced panels was at least a two-bay skin crack with a broken stiffener. A multiple-bay skin crack with broken intermediate stiffeners was conservatively assumed for those design concepts having closely spaced stiffeners (<4 inches). A multiple-bay skin crack was used for the purpose of obtaining a reasonable crack length to facilitate visual inspection.

For those regions that are subjected to extremely low load intensities, the use of fail-safe straps is not required if the structure is capable of supporting limit load with a 20 inch crack, as specified in the fail-safe criteria. Figure 13-12 shows the allowable ultimate design tension stress for this damage condition.

The limit internal loads used in the damage-tolerance analysis were obtained from the combined ultimate internal load calculations which included the effect of air and inertia loads, local pressure loads, and thermal loads. The ultimate stresses for each of the design concepts can be found in the following paragraphs. The limit loads were taken to be two-thirds of the ultimate loads. Only the tensile component acting perpendicular to the crack plane and the shear load were taken into account. For the cases considered in this study, all compressive loads were neglected and thus resulted in a conservative estimation on the residual strength of damaged structure.

Fracture toughness properties of Ti 6Al-4V at room temperature was used in lieu of established data at elevated temperature. A lower bound cut-off value of 0.015 inches on skin thickness was used.

Fail-Safe Analysis - Task I

The Task I results are presented in the following sections according to the different design concepts evaluated. Appropriate comments, conclusions and estimated weight penalties, if any, are also included in the respective sections. Three wing point-design regions were selected for screening all the Task I panel concepts. These regions were point-design regions 41348, 40536, and 40322. The locations of these

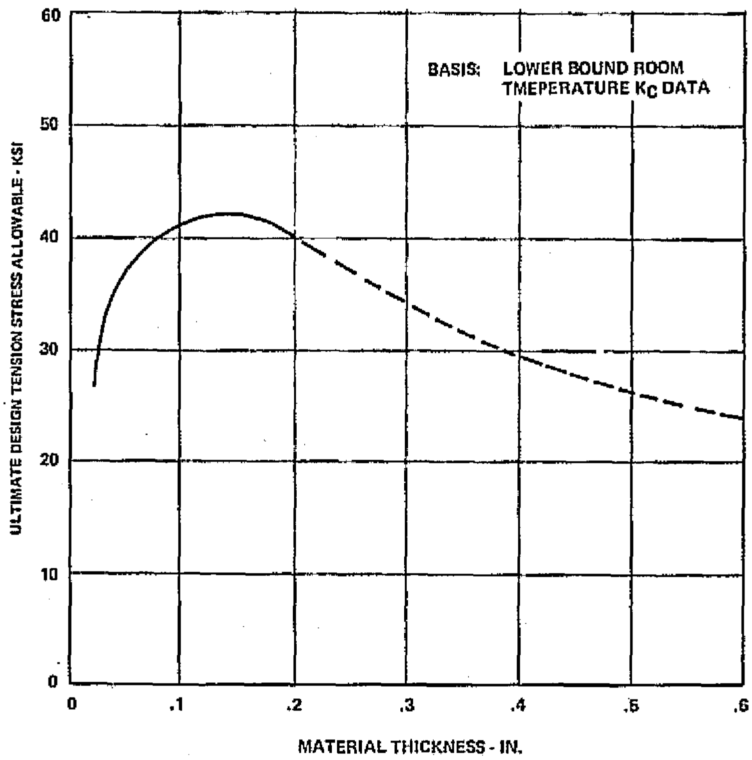


Figure 13-12. Variation in Ultimate Design Tension Stress with Material Thickness with a 20 Inch Crack

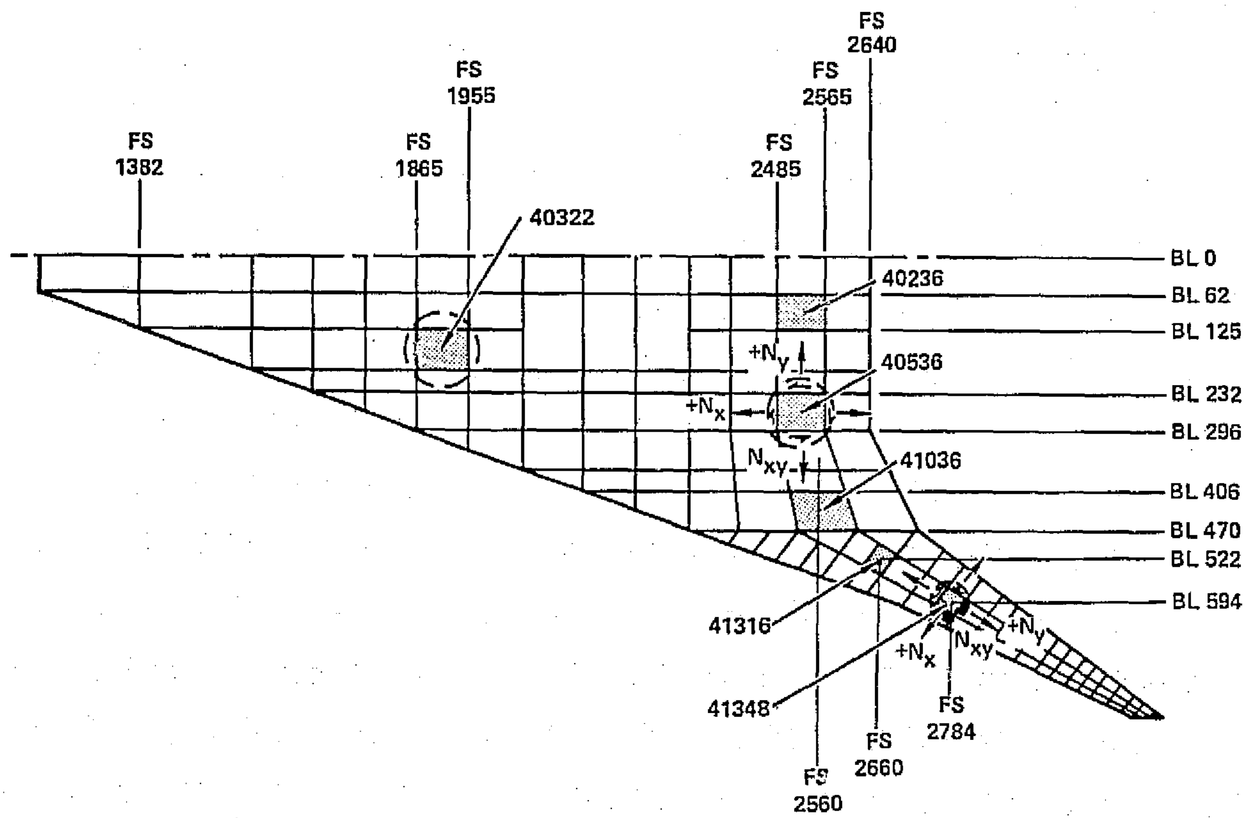


Figure 13-13. Definition of Wing Point Design Regions

point-design regions are as indicated in Figure 13-13. However, only those concepts that appeared to be the most critical were studied. Furthermore, not every possible dimensional variation of a design concept was analyzed. However, an adequate number of possible variations within a particular design concept were selected for analysis based on our experience with this design concept. The representativeness of conclusions reached is thus assured.

In summary, the following comments can be made concerning the damage tolerance of the various design concepts considered. All the chordwise stiffened wing panel concepts meet the fail-safe requirements. However, when the requirement of a broken spar cap is considered, additional structure and its corresponding weight penalty are generally required. For the spanwise stiffened panels, no weight penalty is required for the panels or caps to meet the assumed fail-safe requirements. The composite reinforced spar caps all meet the fail-safe requirements without additional weight penalty. A weight penalty is generally required for the monocoque sandwich panels (honeycomb and truss-core sandwich) except for design regions where low load intensities were indicated. Finally, the metallic fuselage panel concepts considered are fail-safe under the assumed damage condition except at a few isolated panel locations.

Chordwise Stiffened Wing Panels - Four panel concepts were studied within the chordwise stiffened panel arrangement. These concepts are: (1) convex beaded, (2) concave beaded, (3) trapezoidal corrugation-concave beaded, and (4) beaded trapezoidal corrugation-concave beaded. The Structural Concept Analysis Section (Section 12) contains the panel dimensions resulting from the strength analysis. The corresponding skin stress state for these designs, which are used as the basis for the fail-safe investigation, are contained in Table 13-4.

Due to relatively small stiffener spacing of the chordwise stiffened panels, a damage condition of a three-pitch outer skin crack with two broken reinforcing stiffeners (inner bead) was selected. This resulted in a crack size of 5 inches to 13 inches, with the majority of the cracks having a crack length between 7 inches to 10 inches.

The outer skin was treated as a flat panel with the inner beaded skin considered to be the reinforcement. The effective area, A_e, of a reinforcement was taken to be one-third of the total area of the inner-bead between bond lines. The reduction factor of 1/3 is selected based on past experience with various stiffened panel concepts.

TABLE 13-4. SUMMARY OF WING PANEL SKIN STRESSES, CHORDWISE ARRANGEMENT

PANEL CONCEPT	SPAR SPACING (IN)	POINT DESIGN ULTIMATE SKIN STRESS (1) (2) (3)											
		40322				40536				41348			
		UPPER SURFACE		LOWER SURFACE		UPPER SURFACE		LOWER SURFACE		UPPER SURFACE		LOWER SURFACE	
		f_x (ksi)	f_s (ksi)	f_x (ksi)	f_s (ksi)	f_x (ksi)	f_s (ksi)	f_x (ksi)	f_s (ksi)	f_x (ksi)	f_s (ksi)	f_x (ksi)	f_s (ksi)
CONCAVE BEADED 	20	–	3.35	–	3.44	–	43.2	22.0	53.8	–	44.6	24.3	54.3
	30	–	3.04	–	3.14	–	36.4	17.8	44.4	–	37.4	23.5	53.8
	40	–	2.45	–	2.89	–	28.2	15.1	38.6	–	29.4	21.1	48.1
CONVEX BEADED 	20	–	3.81	–	3.85	–	43.9	22.5	54.7	–	45.4	24.2	54.6
	30	–	2.51	–	3.48	–	38.1	19.1	47.4	–	40.5	24.9	55.1
	40	–	2.11	–	2.80	–	34.1	15.5	38.7	–	36.2	20.4	45.0
TRAPEZOIDAL CORRUGATION-CONCAVE BEADED 	20	–	3.06	–	3.80	–	52.3	20.6	50.1	–	61.0	25.0	51.7
	30	–	2.80	–	3.40	–	47.9	17.5	55.4	–	49.7	22.4	52.8
	40	–	2.40	–	3.30	–	40.6	14.6	48.3	–	47.8	19.8	50.7
BEADED TRAP. CORRUGATION-CONCAVE BEADED 	20	–	3.20	–	3.40	–	59.4	21.1	55.3	–	61.2	23.5	52.2
	30	–	2.60	–	3.30	–	53.0	17.3	54.7	–	54.8	21.7	52.0
	40	–	2.40	–	2.90	–	50.0	14.6	55.8	–	52.6	19.0	52.9

(1) ULTIMATE SKIN STRESSES FOR TASK I LOAD CONDITION 31 : 2.5 -g SYMMETRIC MANEUVER AT MACH 1.25

(2) LIMIT STRESS = 2/3 ULTIMATE STRESS

(3) COMPRESSIVE STRESSES CONSERVATIVELY NEGLECTED FOR FAIL SAFE ANALYSIS

Values of fraction toughness were based on the thickness of the outer skin, and were obtained from Figure 5-7 of Reference 6 or Figure 13-10c. A sample panel fail-safe calculation is shown in Table 13-5 for the convex beaded concept at point design region 40536. For visibility, the panel dimensions for this region are shown in Table 13-6.

The damage-tolerance analysis results for the chordwise stiffened panel arrangement are summarized in Table 13-7. No weight penalty was required for any of the chordwise panel concepts analyzed.

In addition to the panel analysis, a fail-safe analysis was also conducted for the case of a broken spar cap. The severity of a broken spar cap is recognized due to the fact that in the chordwise arrangement the spar caps carry the wing spanwise bending loads.

A strength analysis was used to study a basic structural component, as shown in Figure 13-14, in the chordwise stiffened panel arrangement. The top spar cap in Bay 1 was assumed to be broken. A load redistribution study was conducted for the three spar spacings of 20, 30, and 40 inches. The convex beaded panel concept was selected for this analysis as being representative of the chordwise arrangement.

The shear clips, surface panels and spar webs were then resized to carry the resultant loads under the damaged condition. The corresponding weight increase at various spar spacings are presented in Table 13-8. The component weight penalties at point design region 40536 are presented in graphic form in Figure 13-15.

Spanwise Stiffened Wing Panels - Two significantly different groups of design concepts were studied for the Task I spanwise stiffened panel arrangement. The first group consists of panels with separate reinforcements and the second group consists of integrally stiffened panels. The first group (non-integral stiffened) was analyzed using the method described for the chordwise stiffened panels. For the integrally stiffened panels, a damage tolerance penalty is sometimes necessary due to the lack of crack stoppers and a damage condition of complete fracture between manufacturing splices is generally assumed. A discussion on integrally stiffened panels is included in this section.

TABLE 13-5. WING PANEL FAIL-SAFE ANALYSIS - CONVEX BEADED CONCEPT

ITEM	POINT DESIGN REGION 40536					
	UPPER SURFACE			LOWER SURFACE		
SPAR SPACING $L_{P,X}$ IN.	20	30	40	20	30	40
DISTANCE BETWEEN UNBROKEN BEADS (in.)	7.65	8.85	10.65	7.05	8.25	10.65
CRACK LENGTH (in.)	7.65	8.85	10.65	7.05	8.25	10.65
LIMIT STRESSES						
f_x, psi	–	–	–	15000	12700	10300
f_s, psi	29300	25400	22800	36500	31600	25800
EFFECTIVE AREA, (in.^{2}) (1/3 X TOTAL AREA)	0.029	0.048	0.067	0.025	0.035	0.055
SKIN THICKNESS t_u, in.	0.035	0.036	0.040	0.025	0.029	0.037
FRACTURE TOUGHNESS k_o, Ksi – $\sqrt{in.}$	108	109	113	94	100	110
REINFORCEMENT EFFICIENCY, γ	1.26	1.31	1.36	1.29	1.33	1.38
SHEAR CORRECTION FACTOR, ψ	1.00	1.00	1.00	0.38	0.37	0.37
MARGIN OF SAFETY	0.68	0.90	1.08	0.16	0.36	0.68

TABLE 13-6. WING PANEL GEOMETRY - CONVEX BEADED CONCEPT

DESIGN DATA	POINT DESIGN REGION 40536					
	UPPER SURFACE			LOWER SURFACE		
SPAR SPACING $L_{P,X}$ in.	20	30	40	20	30	40
DIMENSIONS						
t_L, in.	.025	.035	.040	.024	.028	.033
t_u, in.	.035	.036	.040	.025	.029	.037
R_L in.	.9	1.1	1.4	.8	1.0	1.4
θ, degrees	87	87	87	87	87	87
b, in.	.75	.75	.75	.75	.75	.75
pitch, in.	2.55	2.95	3.55	2.35	2.75	3.55
WEIGHT DATA						
$\bar{t}$, in.	.070	.085	.097	.058	.068	.084
w, lb/sq. ft.	1.61	1.96	2.24	1.34	1.57	1.94
CRITICAL DESIGN COND.	31	31	31	31	31	31

DIMENSIONS:

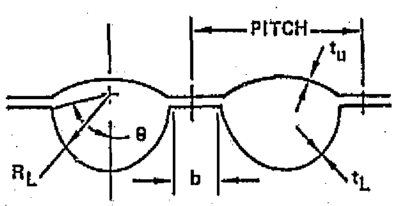


TABLE 13-7. SUMMARY OF WING PANEL FAIL-SAFE ANALYSIS - CHORDWISE ARRANGEMENT

DESIGN CONCEPTS	POINT DESIGN REGION	WING SURFACE	SPAR SPACING (IN.)	CRACK LENGTH, W. (IN.)	$\Sigma Ae/t$ (IN.)	REIN-FORCEMENT EFFICIENCY γ	MARGIN OF SAFETY	WEIGHT PENALTY
CONVEX BEADED	41348	UPPER	20	7.65	1.61	1.26	0.64	NONE
			30	8.85	2.43	1.31	0.79	
			40	10.65	3.10	1.35	0.96	
		LOWER	20	6.45	1.57	1.26	0.23	NONE
			30	7.05	1.43	1.26	0.19	
			40	7.65	1.39	1.27	0.54	
	40536	UPPER	20	7.65	1.66	1.26	0.68	NONE
			30	8.85	2.67	1.31	0.90	
			40	10.65	3.35	1.36	1.08	
		LOWER	20	7.05	2.00	1.29	0.16	NONE
	40322	UPPER	20	7.65	1.87	1.27	HIGH	NONE
			30	9.45	2.43	1.31	HIGH	
		LOWER	20	7.65	1.70	1.26	HIGH	NONE
CONCAVE BEADED	41348	UPPER	20	8.25	2.03	1.28	0.62	NONE
		LOWER	20	6.45	1.45	1.26	0.25	NONE
	40536	UPPER	20	8.25	2.21	1.29	0.64	NONE
		LOWER	20	7.05	2.12	1.30	0.19	NONE
	40322	UPPER	20	7.65	3.13	1.32	HIGH	NONE
TRAPEZOIDAL WITH NO BEAD	41348	UPPER	20	5.25	1.87	1.24	0.32	NONE
		LOWER	20	6.75	1.00	1.24	0.29	NONE
	40536	UPPER	20	6.15	1.87	1.25	0.45	NONE
		LOWER	20	6.75	1.29	1.25	0.42	NONE
	40322	UPPER	20	6.75	2.25	1.27	HIGH	NONE
TRAPEZOIDAL WITH INNER BEAD	41348	LOWER	20	6.45	1.05	1.24	0.34	NONE
	40536	LOWER	20	6.45	1.25	1.25	0.29	NONE
								

NOTE: (1) W = DISTANCE BETWEEN THE TWO UNBROKEN REINFORCEMENTS.
(2) ΣAe = SUM OF THE EFFECTIVE AREAS OF THE TWO INTACT REINFORCEMENTS.
(3) t = SKIN THICKNESS OF THE OUTER SKIN.

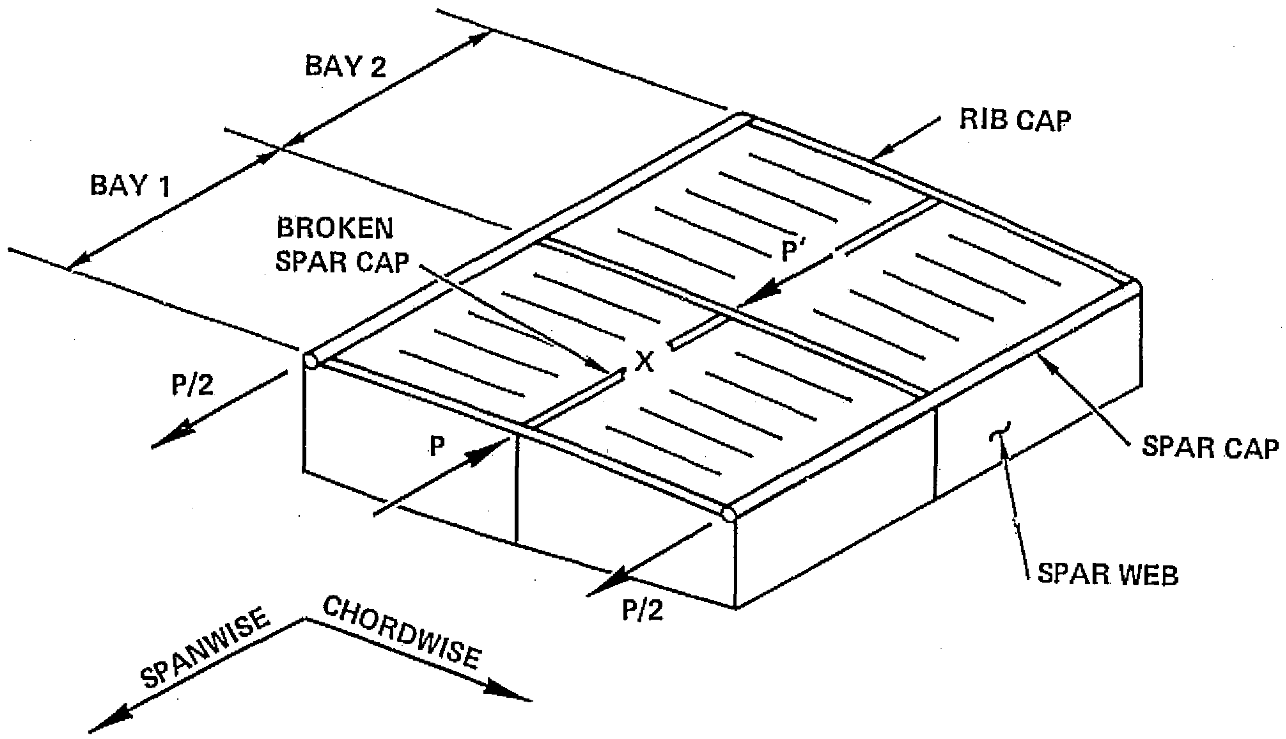


Figure 13-14. Basic Structural Component With a Damaged Spar Cap - Chordwise Arrangement

The structural concepts and detailed panel dimensions are shown in Section 12. The spanwise-stiffened panel concepts considered are: (1) Hat Section stiffened, (2) Zee Section stiffened, (3) Zee Section Integrally Stiffened, and (4) basic Integrally Stiffened.

Non-Integrally Stiffened Designs - For the Hat Section and Zee Section concepts a two-pitch crack with a broken stiffener was used as the damage condition wherever applicable. This assumption, except in one or two cases, resulted in a satisfactory crack length for visual inspection purposes. The effective area, A_e, was calculated using the formula:

$$\Sigma A_e = \frac{A_{st}}{(\bar{y}/\rho)^2 + 1}$$

Where A_{st} is the area of the stringer (or reinforcement), $\bar{y}$ is the distance from the inner surface of the sheet to the centroid of the reinforcement, and

TABLE 13-8. POINT DESIGN WEIGHT PENALTIES FOR A DAMAGED SPAR CAP-CHORDWISE ARRANGEMENT

POINT DESIGN REGION	WEIGHT PENALTY (LB/SQ FT) SPAR SPACING, IN.		
	20	30	40
40536	0.93	0.75	0.63
40322	0.10	0.20	0.27
40236	1.75	1.45	1.38

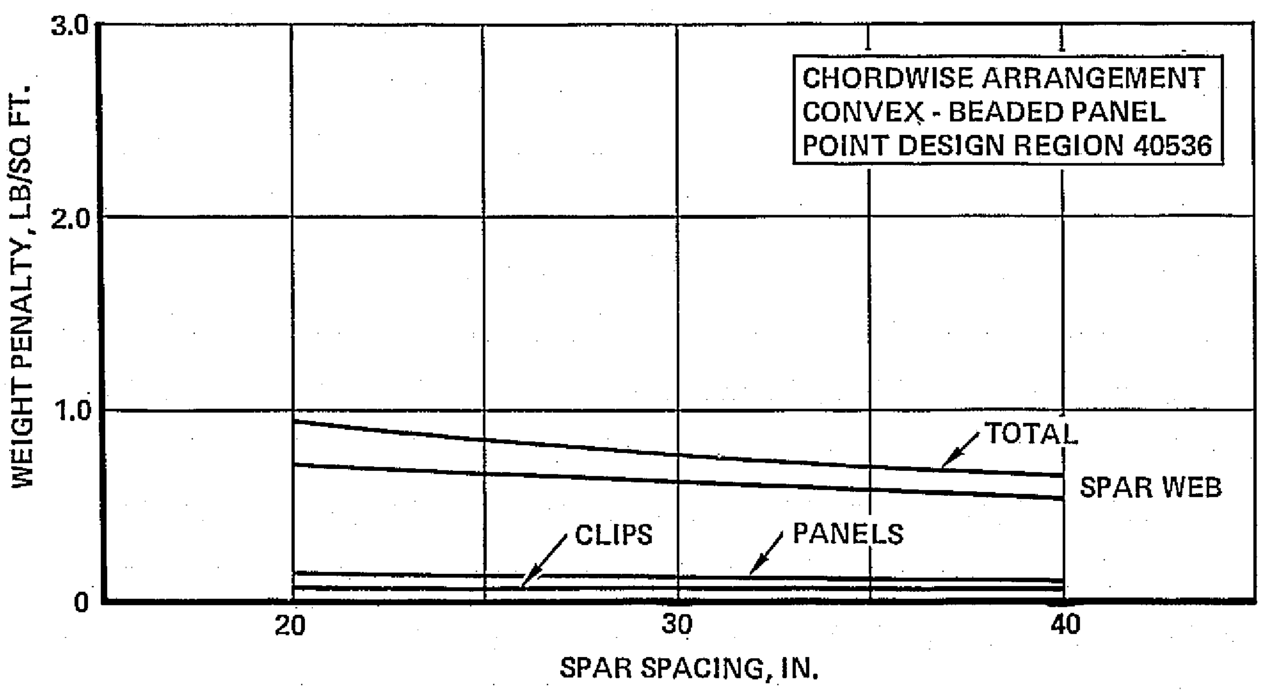


Figure 13-15. Component Weight Penalties for a Damaged Spar Cap-Chordwise Arrangement

ρ is the radius of gyration of the reinforcement. The spar spacing was constant at 60 inches with variable rib spacings of 20 inches, 30 inches, and 40 inches. A sample analysis is shown in Table 13-9 and represents the Hat Section stiffened concept at point design region 40536. The corresponding panel geometry for this region is shown in Table 13-10. The skin stresses at all point design regions for each of the spanwise panel concepts are displayed in Table 13-11.

A summary of the calculations for the spanwise panel concepts with non-integral stiffeners is shown in Table 13-12. No weight penalty is required for these designs.

Integrally Stiffened Designs - The damage condition for integrally stiffened panels was taken as a completely broken panel between skin splices. The fabrication limits for the spanwise integrally stiffened panels allows a maximum width of the extrusion before machining of 22 inches. Since the spar spacing is set at 60 inches, the logical choice for plank spacing is 20 inches (i.e., three planks per bay) with the rib spacing a variable.

Based on the above plank width and rib spacings, a damage-tolerant design can be obtained utilizing the concept of longitudinal-spliced panels, Reference 6. Suitable splices and attachments are used to allow the attachments to transfer the cut load of the broken plank to the two neighboring planks. The two neighboring planks are required to have sufficient effective width so that each panel will support half of the cut load in addition to its normal fail-safe load.

A summary of the fail-safe analysis conducted on the integral stiffened spanwise concepts is shown in Table 13-13. For this analysis, the spar spacing (W) and plank width (W_m) were 60.0 inches and 20.0 inches respectively. An effective width (W_e) of one-third the rib spacing was conservatively selected and the ratio of allowable stress (F_g) to the material ultimate tensile stress (Ftus) was obtained from Figure 4-20 of Reference 6. A positive margin of safety is indicated for each of the critical panel concepts with a minimum margin of +0.24 occurring on the integral stiffened design at a rib spacing of 30 inches. This critical design is located on the wing lower surface at point design region 40322.

TABLE 13-9. WING PANEL FAIL-SAFE ANALYSIS - HAT SECTION STIFFENED CONCEPT

ITEM	POINT DESIGN REGION 40536					
	UPPER SURFACE			LOWER SURFACE		
RIB SPACING $L_{p,y}$, in.	20	30	40	20	30	40
DISTANCE BETWEEN UNBROKEN REIN-FORCEMENTS, in.	6.39	8.42	10.0	8.05	8.96	10.2
CRACK LENGTH, in.	6.39	8.42	10.0	8.05	8.96	10.2
LIMIT STRESSES f_y, psi	–	–	–	41,600	44,600	45,100
f_s, psi	36,200	31,800	27,200	25,500	26,800	26,400
q, lb/in.	2,780	2,781	2,782	2,782	2,782	2,782
EFFECTIVE AREA A_S, in.^{2}	0.374	0.561	0.780	0.667	0.710	0.818
$(\bar{y}/\rho)^2 + 1$	2.772					
A_e, in.^{2}	0.135	0.202	0.281	0.241	0.256	0.295
SKIN THICKNESS, in.	0.077	0.087	0.102	0.109	0.104	0.105
FRACTURE TOUGHNESS k_o, ksi - $\sqrt{\text{in.}}$	132	135	137	138	137	137
REINFORCEMENT EFFICIENCY, γ	1.309	–	–	1.464	–	1.557
ψ	–	–	–	0.815	–	0.826
MARGIN OF SAFETY	1.38	–	–	0.75	–	0.54

TABLE 13-10. WING PANEL GEOMETRY - HAT SECTION STIFFENED CONCEPT

DESIGN DATA	POINT DESIGN REGION 40536					
	UPPER SURFACE			LOWER SURFACE		
RIB SPACING $L_{p,y}$, in.	20	30	40	20	30	40
DIMENSIONS						
t_s, in.	.077	.087	.102	.109	.104	.105
$b_s = b_w = b_z$, in.	1.60	2.11	2.51	2.01	2.24	2.55
$t_w = t_f$, in.	.071	.081	.094	.101	.096	.097
b_f, in.	.479	.605	.753	.600	.672	.765
b_s- b_f, in.	1.12	1.41	1.76	1.40	1.57	1.78
pitch, $b_z + b_s$, in.	3.20	4.22	5.02	4.02	4.48	5.10
WEIGHT DATA						
$\bar{t}$, in.	.194	.221	.258	.275	.262	.266
w, lb/sq. ft.	4.47	5.08	5.94	6.33	6.05	6.12
CRITICAL DESIGN COND.	31	31	31	31	31	31

DIMENSIONS

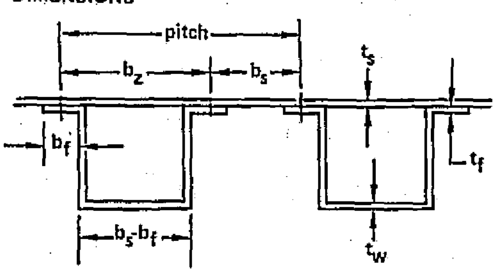
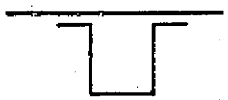
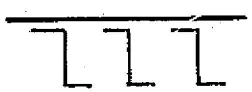


TABLE 13-11. SUMMARY OF WING PANEL SKIN STRESSES, SPANWISE ARRANGEMENT

PANEL CONCEPT	RIB SPACING (IN.)	POINT DESIGN SKIN STRESS (ULTIMATE) (1)(2)(3)											
		40322				40536				41348			
		UPPER SURFACE		LOWER SURFACE		UPPER SURFACE		LOWER SURFACE		UPPER SURFACE		LOWER SURFACE	
		fy (ksi)	f_s (ksi)	f_y (ksi)	f_s (ksi)	f_y (ksi)	f_s (ksi)	f_y (ksi)	f_s (ksi)	f_y (ksi)	f_s (ksi)	f_y (ksi)	f_s (ksi)
HAT SECTION	20	–	11.1	46.6	14.3	–	54.2	62.4	38.3	–	51.2	61.1	43.8
	30	–	8.2	65.0	14.3	–	47.7	66.9	40.1	–	40.9	60.3	43.4
	40	–	6.6	84.0	13.9	–	40.9	67.7	39.6	–	34.7	60.9	44.0
ZEE SECTION	20	–	9.2	48.4	14.5	–	45.1	68.4	41.1	–	45.4	61.4	43.2
	30	–	6.7	66.3	14.3	–	41.5	68.3	40.2	–	36.4	60.7	42.9
	40	–	5.2	87.2	14.1	–	36.6	69.3	39.8	–	31.0	60.8	43.1
INTEGRAL STIFFENED	20	–	7.4	56.2	14.2	–	41.5	63.6	37.4	–	37.9	55.9	38.7
	30	–	5.3	81.8	14.2	–	35.3	63.6	36.4	–	30.0	56.1	39.2
	40	–	4.2	81.5	11.3	–	30.0	63.6	35.4	–	25.3	56.5	39.6
INTEGRAL ZEE	20	–	9.4	54.7	14.4	–	43.1	61.6	34.4	–	45.3	58.6	38.3
	30	–	6.8	77.6	14.4	–	40.6	62.5	34.2	–	36.7	59.7	39.1
	40	–	5.4	83.5	12.2	–	36.7	65.9	35.1	–	31.1	59.7	39.3

(1) ULTIMATE SKIN STRESSES FOR TASK I LOAD CONDITION 31 : 2.5-g SYMMETRIC MANEUVER AT MACH 1.25
(2) LIMIT STRESS = 2/3 ULTIMATE STRESS
(3) COMPRESSIVE STRESSES CONSERVATIVELY NEGLECTED FOR PANEL FAIL SAFE ANALYSIS

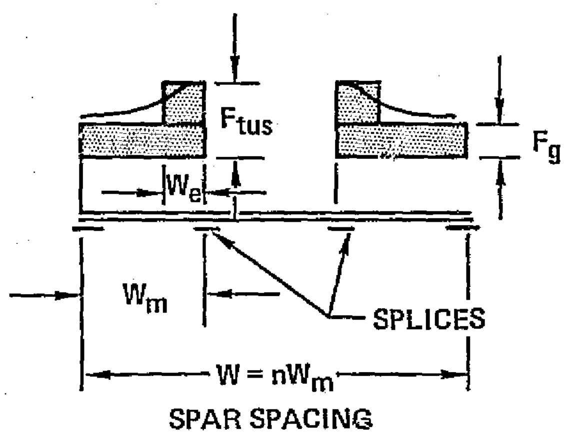
TABLE 13-12. SUMMARY OF WING PANEL FAIL-SAFE ANALYSES - SPANWISE ARRANGEMENT, NON-INTEGRAL STIFFENED DESIGNS

DESIGN CONCEPTS	POINT DESIGN REGION	WING SURFACE	SPAR SPACING (IN.)	CRACK LENGTH, W (IN.)	ΣAe/t (IN.)	REINFORCE-MENT EFFICIENCY	MARGIN OF SAFETY	WEIGHT PENALTY
HAT SECTION STIFFENED 	41348	UPPER	20	5.15	2.81	1.27	1.51	NONE
		LOWER	20	5.62	3.08	1.35	0.75	NONE
			40	7.92	4.35	1.45	0.57	
	40536	UPPER	20	6.39	3.51	1.31	1.38	NONE
		LOWER	20	8.05	4.43	1.46	0.75	NONE
			40	10.2	5.60	1.56	0.54	
	40322	LOWER	20	3.16	1.80	1.27	1.26	NONE
			40	4.53	2.48	1.34	0.16	
ZEE SECTION STIFFENED 	41348	LOWER	20	6.52	1.75	1.31	0.57	NONE
			40	9.16	2.47	1.39	0.42	
	40536	UPPER	20	8.27	2.24	1.29	1.56	NONE
		LOWER	20	8.78	2.39	1.39	1.48	NONE
			40	11.7	3.18	0.45	0.33	
	40322	LOWER	20	3.63	1.00	1.23	1.29	NONE
			40	5.20	1.33	0.98	0.01	

NOTES:
(1) W = DISTANCE BETWEEN TWO UNBROKEN STIFFENERS, ALSO CRACK LENGTH
(2) ΣA_e = EFFECTIVE AREA OF THE TWO UNBROKEN STIFFENERS
(3) t = SKIN THICKNESS

TABLE 13-13. SUMMARY OF WING PANEL FAIL-SAFE ANALYSES - SPANWISE ARRANGEMENT, INTEGRAL STIFFENED DESIGNS

SPAR SPACING W (IN.)	PLANK WIDTH W_m (IN.)	NO. OF PLANKS n (IN.)	RIB SPACING $L=L_T$ (IN.)	EFF WIDTH, W_e (IN.)	β $\left(\frac{2W_e}{W}\right)$	$\frac{F_g}{F_{tus}}$	F_{tus} (KSi)	ALLOWABLE GROSS AREA STRESS, KSi: LIMIT F_g	ALLOWABLE GROSS AREA STRESS, KSi: ULT. F_g	MAX. APPLIED STRESS, ULT. f_y (Ksi)	MIN. MARGIN OF SAFETY MS
60.0	20.0	3	20	6.67	.222	.40	135	54.0	81.0	63.6	+ 0.27
			30	10.00	.333	.50	135	67.5	101.2	81.8	+ 0.24
			40	13.33	.444	.57	135	77.0	115.5	83.5	+ 0.38



NOTES:

(1) ANALYSIS REFERENCE 6.

(2) NOMENCLATURE:
L_T = LENGTH REQUIRED FOR SPLICE ATTACHMENTS TO TRANSFER THE LOAD; $L_T = L$
W_e = EFFECTIVE WIDTH; $W_e = L_T/3$
β = EFFECTIVE WIDTH PARAMETER, $\beta = 2W_e/W$
$MS = F_g/f_y - 1$
F_{tus} = MATERIAL ULTIMATE TENSION STRENGTH

(3) MAXIMUM APPLIED STRESSES FOR THE INTEGRALLY STIFFENED DESIGNS OCCUR ON THE LOWER WING SURFACE AT POINT DESIGN REGIONS 40322 AND 40536, SEE TABLE 13-9.

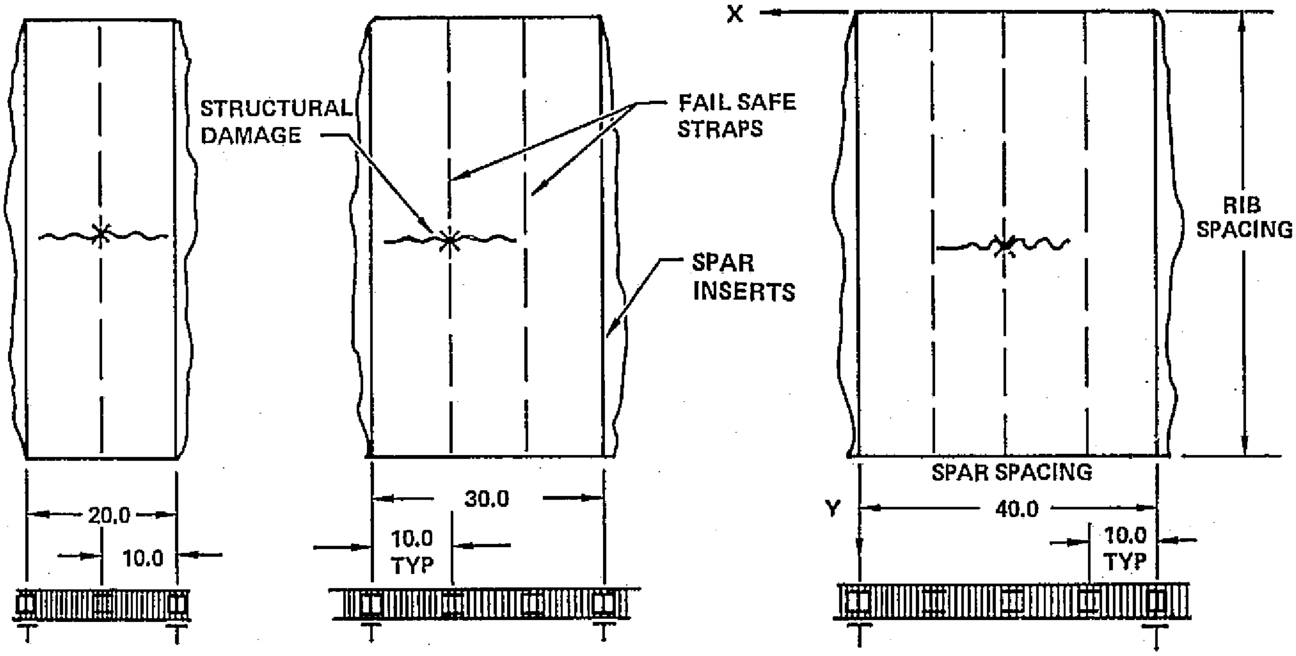
The splice and fastener system were selected based on the following considerations. The splice was selected with a thickness equal to or at most twice the skin thickness and the fastener size and spacing selected such that an effective width of 1/3 W_m was obtained. Also the joint was designed to be bearing critical rather than shear critical. Finally, the usual precautions pertaining to fatigue requirements of splice and attachment were observed. The structural integrity of the joint design selected should be substantiated by a fail-safe test conducted on a multi-bay panel with one plank broken.

An analysis similar to the case of the chordwise stiffened arrangement was conducted for the damage condition of a broken rib cap. However, due to the relatively small section of the rib cap and its direct attachment to the skin, this damage condition was not critical for limit load. Consequently there is no weight penalty.

Monocoque Wing Panels - The honeycomb sandwich and truss-core sandwich panel concepts were investigated during the Task I effort. Two types of panel inserts were considered for the monocoque panel concepts; metallic and densified honeycomb inserts. However, only densified core inserts were considered for the truss-core concept. See Section 1, Structural Design Concepts, for descriptions of these concepts and their close-out design.

As in the cases of other design concepts studied, three point design regions (41348, 40322, 40536) were selected for preliminary screening purposes. For honeycomb sandwich panels, the rib spacing was kept constant at 60 inches with the exception of point design region 40322 where a 130 inch rib spacing was used. For the truss-core sandwich panels, core orientations in both the spanwise and chordwise direction were considered. The spanwise core orientation proved to be the most efficient for point design region 41348 and 40536 with a constant rib spacing of 60 inches, spar spacing being a variable. The chordwise core orientation was used at point design region 40322 with a constant rib spacing of 130 inches.

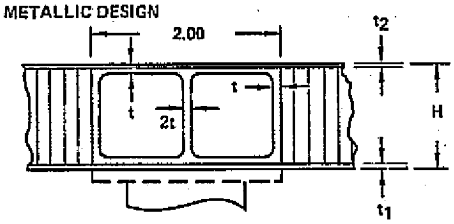
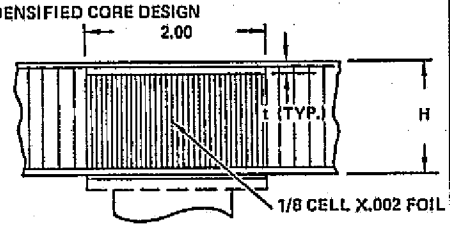
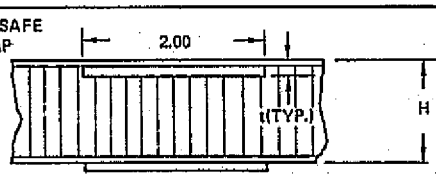
The assumed damage condition was a two-bay crack with a broken reinforcement (i.e., insert or panel closeout member) and both face skins damaged. See Figure 13-16 for a graphic display of the damaged condition.



DAMAGE CONDITION:

- 2 BAY SKIN CRACK (UPPER AND LOWER SKINS)
- BROKEN INSERT OR STRAP

Figure 13-16. Monocoque Panel Damage Configurations

REINFORCEMENT DESIGN	REINFORCEMENT AREA (A_R), IN^2	EFFECTIVE AREA (A_e), IN^2
SPAR INSERTS • METALLIC DESIGN  	GENERAL EQUATION: $A_R = 4t\left[H-(t_2+t_1)\right]+4t$ (1) MINIMUM DESIGN REQUIREMENTS: SPAR SPACING (IN.) / THICKNESS t (IN.) / A_R (IN.^{2}) 20 / .040 / 0.181 30 / .040 / 0.209 40 / .040 / 0.262 WEB CAP NEGLECTED	$A_e = 0.85\ A_R$
• DENSIFIED CORE DESIGN  	GENERAL EQUATION: $A_R = 4t$ (1) MINIMUM DESIGN REQUIREMENTS: SPAR SPACING (IN.) / THICKNESS t (IN.) / A_R (IN^2) 20 / .022 / 0.088 30 / .022 / 0.088 40 / .025 / 0.100 DENSIFIED CORE AND WEB CAP NEGLECTED	$A_e = A_R$
FAIL SAFE STRAP  	GENERAL EQUATION: $A_R = 4t$ CORE NEGLECTED	$A_e = A_R$

(1) MINIMUM DESIGN INSERT REQUIREMENTS FOR POINT DESIGN REGION 40536, LOWER SURFACE.

Figure 13-17. Honeycomb Sandwich Insert Geometry and Data

The broken reinforcement can either be an insert or a fail-safe strap. Effective area (A_e) of the metallic inserts is assumed to be 85 percent of the total insert area. In addition, all doublers were considered to be fully effective. For densified honeycomb inserts, the core was considered to be ineffective and only the area of the doublers was considered for the fail-safe analysis.

Figure 13-17 shows the type of inserts and the effective area equations used in the fail-safe analysis of the honeycomb sandwich panels.

Since both face skins are damaged, only the in-plane membrane loads existing on the damage area are considered redistributed to the adjacent structure. Table 13-14 contains a summary of inplane stresses (ultimate) for the monocoque panel concepts.

Generally, the outer and the inner skins have similar thicknesses. One exception is for the honeycomb sandwich panel arrangement with metallic inserts at point design region 41348. The skin thickness for the outer skin is 0.020 inches while the inner skin has a thickness of 0.070 inches. In all cases, the sum of the thicknesses of the outer and the inner skins (t_1 + t_2 are taken to be the skin thickness (t) in calculating the ratio of $\Sigma Ae/t$. Critical fracture toughness values were taken to be the lowest between that of the outer and the inner skin. Table 13-15 contains the honeycomb sandwich panel geometry at point design region 40536. In addition, the fail-safe analyses for the panels at this region are shown in Table 13-16. For this analysis the required strap and spar areas have been defined to obtain a zero margin of safety. The corresponding weight penalty for the honeycomb sandwich concept at point design region 40536 are shown in Table 13-17. The weight penalties associated with both the metal and densified core inserts are presented.

A summary of the monocoque panel fail-safe results is presented in Tables 13-18 and 13-19 for the honeycomb sandwich and truss-core sandwich concepts respectively. In general, a weight penalty is required for all lower surface wing panels at the selected point design regions. One notable exception is the honeycomb sandwich design (Table 13-18) at point design region 40322. The design loads at this region are low and the structure is capable of withstanding a 20 inch crack without failure. The corresponding area for the truss-core concept exhibits a small weight penalty.

TABLE 13-14. SUMMARY OF WING PANEL SKIN STRESSES, MONOCOQUE ARRANGEMENT

PANEL CONCEPT	SPAR SPACING (IN.)	POINT DESIGN SKIN STRESS (ULT.) – ksi (1)								
		40322 (2)			40536 (2)			41348 (2)		
		LOWER SURFACE			LOWER SURFACE			LOWER SURFACE		
		f_x	f_y	f_s	f_x	f_y	f_s	f_x	f_y	f_s
HONEYCOMB SANDWICH	20	18.6	28.4	4.64	25.5	81.3	33.7	16.6	76.1	36.2
	30	18.5	26.3	4.18	24.4	80.2	33.3	15.8	76.8	36.5
	40	18.7	26.1	4.21	23.9	79.9	33.2	14.8	75.2	35.7
TRUSS-CORE SANDWICH	20	23.9	37.8	4.46	26.2	80.3	34.4	20.0	77.1	39.2
	30	26.3	43.2	4.32	25.8	81.0	34.0	17.3	77.5	38.5
	40	17.9	46.4	3.99	25.7	81.0	33.8	17.0	78.2	38.0

(1) LIMIT STRESS = 2/3 ULTIMATE STRESS

(2) CRITICAL DESIGN CONDITION:
POINT DESIGN REGION 40322 – CONDITION 20, START-OF CRUISE.
POINT DESIGN REGION 40536 AND 41348 – CONDITION 31, 2.5-g SYMMETRIC MANEUVER AT M1.25

TABLE 13-15. WING PANEL GEOMETRY - HONEYCOMB SANDWICH CONCEPT

DESIGN DATA	POINT DESIGN REGION 40536					
	UPPER SURFACE			LOWER SURFACE		
SPACING, IN.						
RIB	60	60	60	60	60	60
SPAR	20	30	40	20	30	40
DIMENSIONS						
H, IN.	.837	1.27	1.48	.290	.454	.781
t_1, IN.	.053	.052	.050	.076	.076	.087
t_2, IN.	.052	.051	.050	.061	.063	.053
t_c, IN.	.002	.002	.002	.002	.002	.002
S, IN.	.258	.185	.167	.500	.500	.500
WEIGHT DATA						
$\bar{t}$, IN.	.117	.131	.139	.139	.142	.145
W, LB./SQ. FT.	2.69	3.02	3.21	3.20	3.28	3.35
CRITICAL DESIGN COND.	31	31	31	31	31	31

DIMENSIONS

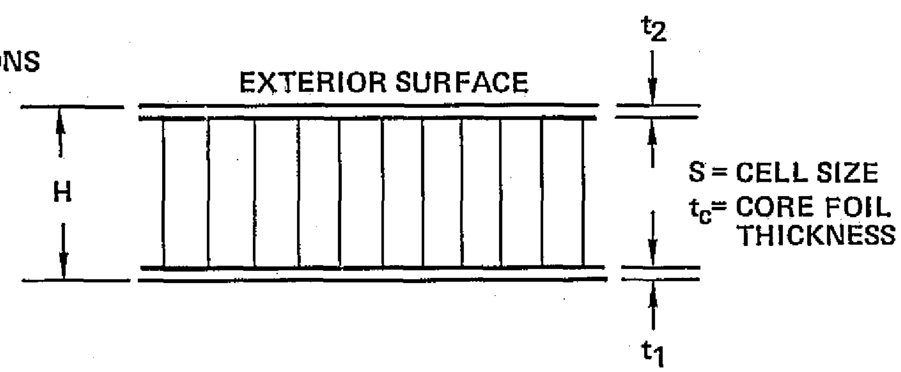


TABLE 13-16. WING PANEL FAIL-SAFE ANALYSIS - HONEYCOMB SANDWICH CONCEPT

ITEM	POINT DESIGN REGION 40536		
	LOWER SURFACE		
SPAR SPACING, IN.	20	30	40
PANEL DIMENSIONS			
t_2, IN.	.061	.063	.053
t_1, IN.	.076	.076	.087
H, IN.	.290	.454	.781
REINFORCEMENT AREAS (A_R)			
SPAR CAP, IN^2	.862	.785	.922
STRAP, IN^2	.733	.667	.784
EFFECTIVE AREA			
Ae, IN^2	.733	.667	.784
LIMIT STRESSES			
COND. NO.	31	31	31
f_y, psi	54,200	53,500	53,300
f_s, psi	22,500	22,200	22,100
FRACTURE TOUGHNESS k_o, Ksi - $\sqrt{in.}$			
k_o (t_1)	132	132	136
k_o (t_2)	126	128	122
CRACK LENGTH (L), in.	20	20	20
REINFORCEMENT EFFICIENCY			
γ	2.16	2.10	2.19
ψ	0.89	0.89	0.89
MARGIN OF SAFETY	0.00	0.00	0.00

TABLE 13-17. WING PANEL WEIGHT PENALTY - HONEYCOMB SANDWICH CONCEPT

POINT DESIGN REGION 40536		MINIMUM DESIGN REQUIRE-MENTS	FAIL SAFE(1) REQUIREMENTS				
DESIGN CONCEPTS	SPAR SPACING, b (IN.)	A_{SPAR} (IN^2)	A_{SPAR} (IN^2)	A_{STRAP} (IN^2)	NO. STRAPS n	CRACK WIDTH (IN.)	WEIGHT(2) PENALTY, ΔW (LB./SQ. FT.)
HONEYCOMB SANDWICH METAL INSERTS	20	.181	.862	.733	1	20	1.63
	30	.209	.785	.667	2	20	1.47
	40	.262	.922	.784	3	20	1.73
HONEYCOMB SANDWICH DENSIFIED CORE	20	.088	.733	.733	1	20	1.59
	30	.088	.667	.667	2	20	1.47
	40	.100	.784	.784	3	20	1.75

NOTES:

(1) A_{SPAR} IS 85% EFFECTIVE FOR THE METALLIC INSERT DESIGN; 100% EFFECTIVE FOR THE DENSIFIED CORE DESIGN.

(2) WEIGHT PENALTY EQUATION (EQUIVALENT PANEL WEIGHT):

$$\Delta W = 144\rho\left[(A_{SPAR,\,FS} - A_{SPAR,\,MIN}) + nA_{STRAP}\right]/b$$

WHERE:

ρ = MATERIAL DENSITY, lb./in^3

$A_{SPAR,FS}$ = SPAR CAP AREA REQUIRED FOR FAIL SAFE

$A_{SPAR,MIN}$ = SPAR CAP AREA MINIMUM DESIGN REQUIREMENT

A_{STRAP} = STRAP AREA REQUIRED FOR FAIL SAFE

n = NUMBER OF STRAPS REQUIRED

b = SPAR SPACING

TABLE 13-18. SUMMARY OF WING PANEL FAIL-SAFE ANALYSES - HONEYCOMB SANDWICH CONCEPT

DESIGN CONCEPTS	POINT DESIGN REGION	WING SURFACE	SPAR SPACING (IN.)	CRACK LENGTH (IN.)	$\Sigma Ae/t$ (1)(2) (IN.)	REINFORCEMENT EFFICIENCIES		MARGIN OF SAFETY	WEIGHT (3) PENALTY (LB/SQ FT)
						γ	ψ		
HONEYCOMB SANDWICH, METALLIC INSERTS	40536	LOWER	20	20	10.70	2.16	0.89	+0.00	1.63
			30	20	9.60	2.10	0.89	+0.00	1.47
			40	20	11.20	2.19	0.89	+0.00	1.73
	40322	LOWER	20	20	15.81	2.00^{+}	0.98	+0.31	NONE
			40	20	23.27	2.00^{+}	0.98	+0.01	NONE
	41348	LOWER	20	20	12.74	2.00^{+}	0.87	+0.00	1.24
			40	20	13.66	2.00^{+}	0.87	+0.00	1.41
HONEYCOMB SANDWICH, DENSIFIED CORE INSERTS	40536	LOWER	20	20	10.70	2.16	0.89	+0.00	1.59
			30	20	9.60	2.10	0.89	+0.00	1.47
			40	20	11.20	2.19	0.89	+0.00	1.75
	40322	LOWER	20	20	15.81	2.00^{+}	0.98	+0.31	NONE
			40	20	23.27	2.00^{+}	0.98	+0.01	NONE
	41348	LOWER	20	20	12.74	2.00^{+}	0.87	+0.00	1.23
			40	20	13.66	2.00^{+}	0.87	+0.00	1.42

(1) ΣAe = SUM OF THE EFFECTIVE AREAS

(2) t = SUM OF THE FACE SHEET THICKNESSES.

(3) WEIGHT PENALTY (STRAPS AND/OR ADDED SPAR CAP AREA) TRANSLATED INTO EQUIVALENT SURFACE PANEL WEIGHT

TABLE 13-19. SUMMARY OF WING PANEL FAIL-SAFE ANALYSES - TRUSS-CORE SANDWICH CONCEPT

DESIGN CONCEPT	POINT DESIGN REGION	WING SURFACE	SPAR SPACING (IN.)	CRACK LENGTH (IN.)	(1)(2) $\Sigma Ae/t$ (IN.)	REINFORCEMENT EFFICIENCIES γ	REINFORCEMENT EFFICIENCIES ψ	MARGIN OF SAFETY	(3) WEIGHT PENALTY (LB./SQ. FT.)
TRUSS-CORE SANDWICH DENSIFIED CORE INSERTS	40322	LOWER	20	20	1.40	1.45	0.97	+0.00	0.03
			30	20	2.90	1.63	0.99	+0.00	0.04
			40	20	3.30	1.67	0.99	+0.00	0.04
	40536	LOWER	20	20.2	9.75	2.11	0.89	+0.00	1.36
			30	19.8	9.50	2.09	0.89	+0.00	1.43
			40	19.9	9.35	2.08	0.89	+0.00	1.45
	41348	LOWER	20	20.4	15.00	2.54	0.85	+0.00	1.24
			30	20.6	15.00	2.57	0.86	+0.00	1.34
			40	19.6	15.00	2.54	0.86	+0.00	1.35

(1) Σ Ae = SUM OF THE EFFECTIVE AREAS

(2) t = SUM OF THE FACE SHEET THICKNESSES

(3) WEIGHT PENALTY (STRAPS AND/OR ADDED SPAR CAP AREA) TRANSLATED INTO EQUIVALENT SURFACE PANEL WEIGHT.

In conclusion, sizeable weight penalties are required for both concepts in the highly loaded regions of the aft box and wing tip.

Composite Reinforced Spar Cap - Composite reinforced spar caps were studied for possible weight saving advantages. From the point of view of damage-tolerance design the multiple element characteristics (i.e., load redistributed to remaining undamaged elements) of the composite reinforced spar cap is very attractive. However, when other factors such as manufacturing considerations, the possibility of debonding due to shear deformation, and eccentricity due to a broken member are taken into account, the choice becomes less clear-cut. Obviously, a more detailed study is necessary before making a final selection. However, a simplified strength analysis was conducted to indicate the damage tolerance trends for this type of design.

The detail dimensions of the Boron/polyimide reinforced spar caps are shown in Table 13-20 for the lower surface caps at point design regions 40322, 40536, and 41348.

The ultimate load carried in each individual metal or composite element, as well as the total ultimate load, was calculated at the above point design regions for the spar spacing values of 20 and 40 inches. The exception is the lower surface spar caps for 20 inch spacing at point design region 40322. Composite reinforcement was not used in this region due to the negligible weight saving indicated over the homogeneous metal design. The fail-safe analysis results are presented in Table 13-21. In review, all composite reinforced spar caps are damage-tolerant under the damage condition of a single broken element. The exception being the slightly negative margin (1-percent) indicated for the spar caps with 40 inches spacing at point design region 40322. No redesign of these caps was attempted since the strength analysis indicated the smaller spar spacings, between 20 and 30 inches, were also the least weight designs.

Fuselage Analysis - The Task I design effort consisted of two phases; first, an initial screening phase where the fuselage panel candidates were subjected to structural analysis to select the most promising concept(s); then an in-depth structural analysis of this (these) concept(s) to provide a sound basis for the fuselage detail design studies of Task IIB.

TABLE 13-20. SPAR CAP GEOMETRY - COMPOSITE REINFORCED CONCEPT

POINT DESIGN REGION	SPAR SPACING (IN.)	SPAR CAP DIMENSIONS					
		h (IN.)	b (IN.)	H (IN.)	W (IN.)	t_1 (IN.)	t_2 (IN.)
40322 LOWER SURFACE	20(1)	–	–	–	1.50	.21	.05
	40	.12	.50	1.00	1.50	.09	.11
40536 LOWER SURFACE	20	.49	1.00	1.20	2.50	.12	.13
	40	.67	1.75	1.20	4.00	.08	.13
41348 LOWER SURFACE	20	.42	1.00	1.20	2.50	.12	.13
	40	.63	1.75	1.20	4.00	.08	.13

(1) NO APPRECIABLE WEIGHT SAVINGS INDICATED, USED ALL METAL SPAR CAPS.

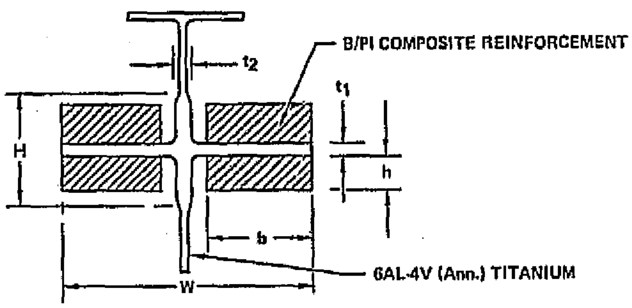


TABLE 13-21. SUMMARY OF SPAR CAP FAIL-SAFE ANALYSES - COMPOSITE REINFORCED CONCEPT

POINT(1) DESIGN REGION	SPAR SPACING (IN.)	TOTAL(1)(2) APPLIED LOAD, KIPS		ALLOWABLE(3) MEMBER LOAD, KIPS		DAMAGED CAP(4) ALLOWABLE LOAD (P_A), KIPS		MARGIN(5) OF SAFETY
		$P_{ULT.}$	P_{LIMIT}	METAL ELEMENT P_M	COMPOSITE ELEMENT P_C	BROKEN METAL ELEMENT	BROKEN COMPOSITE ELEMENT	
40322	40	52	34.7	17.6	8.6	34.4	43.4	.01
40536	20	315	210	35.0	70.0	280.0	245.0	+.17
	40	673	449	33.0	160.0	640.0	513.0	+.14
41348	20	271	180	35.0	59.0	236.0	212.0	+.18
	40	633	422	33.0	150.0	600.0	483.0	+.14

(1) LOWER SURFACE SPAR CAPS, MAXIMUM TENSION LOADS.

(2) LIMIT LOAD = $\frac{2}{3}$ X ULTIMATE LOAD

(3) COMPOSITE ELEMENT LOAD ARE UNIT VALUES, TOTAL LOAD SUSTAINED BY THE COMPOSITE ELEMENTS IS FOUR TIMES THE UNIT VALUES.

(4) DAMAGED CAP ALLOWABLES:

BROKEN METAL MEMBER: $P_A = 4 \times P_C$

BROKEN COMPOSITE MEMBER: $P_A = P_M + 3 \times P_C$

(5) MARGIN OF SAFETY = $\frac{\text{MIN. } P_A}{P_{LIMIT}} - 1$

Fail-safe analyses were conducted during each of the above phases to provide credence to the selection procedure used to define the most promising fuselage panel candidate. The analytical method described in Figure 13-10 was used with two types of cracks being considered: (1) circumferential cracks, and (2) longitudinal cracks. For these two types of cracks in the fuselage, the methods of analysis for flat shell structure apply with the exception of a 50 percent reduction in fracture toughness (k_o) imposed on the cases involving longitudinal cracks. This reduction is based on a conservative estimation of the effects of curvature of the fuselage panels.

For circumferential cracks, a damage condition of a two-bay crack with one broken stringer was considered. The corresponding damage condition for the longitudinal cracks was a two-bay crack with the intermediate frame broken, i.e., a 40 inch crack for a fuselage frame spacing of 20 inches.

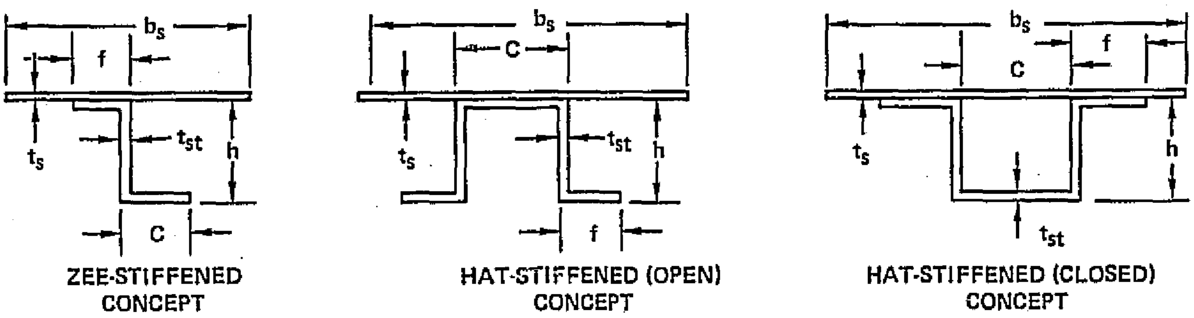
The fuselage panel concepts analyzed in support of the initial screening effort were the zee-stiffened concept, open hat-stiffened concept, and the closed hat-stiffened concept. These concepts and their corresponding dimensions for the maximum tension case (top centerline panels) are shown in Table 13-22. These dimensions were defined from the strength analysis (Section 12).

A total of four fuselage stations were examined for damage-tolerance analysis, i.e., FS 750, FS 2000, FS 2500, and FS 3000 as indicated on Figure 13-18. Different locations at each station were also examined. The fuselage maximum tension stresses are shown in Table 13-23 for each of the concepts investigated in the initial screening. Review of these stresses indicate areas that were designed up to the allowable ultimate design gross area stress (90,000 psi) commensurate with the life and assumed fatigue quality of the fuselage.

A sample of the fuselage fail-safe analysis is shown in Table 13-24. This analysis was conducted for each of the three candidate concepts at FS 2500. The panel geometry and applied stress state reflect the top centerline panel location. Positive margins of safety are indicated for each design for both the longitudinal and circumferential crack damage conditions. The lowest margins of safety are associated with the circumferential crack condition, with the minimum value (positive 3-percent) occurring on the closed hat concept.

TABLE 13-22. FUSELAGE PANEL GEOMETRY-INITIAL SCREENING

POINT DESIGN REGION	LOCATION	PANEL CONCEPT	FUSELAGE PANEL DIMENSION						
			b_s (IN.)	t_s (IN.)	C (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	$\bar{t}$ (IN.)
FS 2000 AND FS 3000	TOP	ZEE STIFF	4.00	.100	0.75	1.00	1.25	0.90	.160
		OPEN HAT	5.00	.080	1.25	.80	1.25	.063	.151
		CLOSED HAT	6.00	.080	1.50	.80	1.25	.063	.142
FS 2500	TOP	ZEE STIFF	4.00	.100	0.75	1.00	1.25	.110	.190
		OPEN HAT	5.00	.080	1.25	.80	1.25	.070	.159
		CLOSED HAT	6.00	.080	1.50	.80	1.25	.080	.159



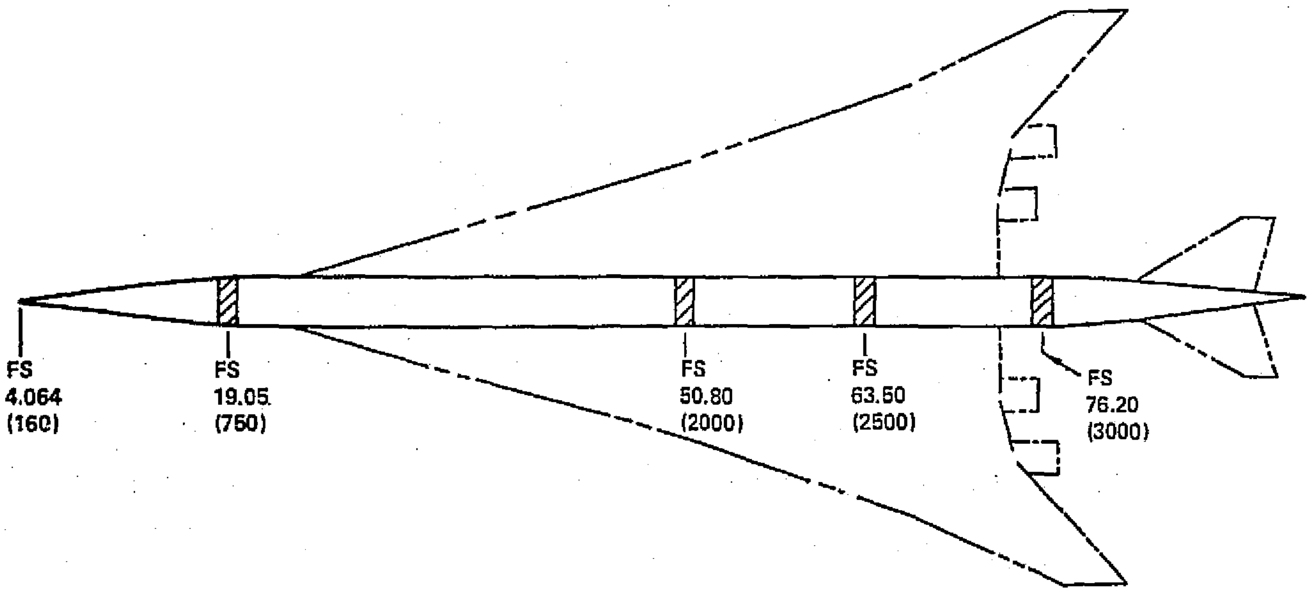


Figure 13-18. Definition of Fuselage Point Design Regions

TABLE 13-23. SUMMARY OF FUSELAGE PANEL STRESSES, INITIAL SCREENING

LOCATION	FUSELAGE PANEL CONCEPTS	FUSELAGE SKIN STRESSES (ULT) – KSI[1]					
		FS 2000 AND FS 3000			FS 2500		
		f_x	f_θ	f_{xy}	f_x	f_θ	f_{xy}
TOP ℄ PANELS (MAX. TENSION)	ZEE STIFF.	72.4	12.0	–	82.6	12.0	–
	HAT STIFF. (OPEN)	77.3	15.0	–	90.0	15.0	–
	HAT STIFF. (CLOSED)	81.3	15.0	–	90.0	15.0	–

(1) LIMIT STRESS = 2/3 ULTIMATE STRESS

(2) LOAD CONDITION: START-OF-CRUISE

TABLE 13-24. FUSELAGE PANEL FAIL-SAFE ANALYSIS, INITIAL SCREENING

ITEM	POINT DESIGN REGION – FS 2500					
	ZEE STIFF		OPEN HAT		CLOSED HAT	
CRACK TYPE	LONG.	CIRCUM.	LONG.	CIRCUM.	LONG.	CIRCUM.
LOCATION	TOP ℄ PANEL		TOP ℄ PANEL		TOP ℄ PANEL	
PANEL GEOMETRY						
t_s, in.	.10	.10	.08	.08	.08	.08
b_s, in.	–	4.00	–	5.00	–	6.00
FRAME SPACING, IN.	20.0	–	20.0	–	20.0	–
REINF. PROP.						
AREA (A_θ), in^2	.072	.130	.072	.128	.072	170
$\Sigma A_\theta/t_s$, in.	1.44	2.60	1.80	3.20	1.80	4.25
LIMIT STRESSES[2]						
f_x, ksi	55.1	55.1	60.0	60.0	60.0	60.0
f_θ, ksi	8.0	8.0	10.0	10.0	10.0	10.0
f_{xy}, ksi	–	–	–	–	–	–
FRACTURE TOUGH.						
k_0, ksi $-\sqrt{in.}$	138	138	135	135	135	135
1/2 k_0, ksi $-\sqrt{in.}$	69	–	67.5	–	67.5	–
CRACK LENGTH						
L, in.	40.0	8.0	40.0	10.0	40.0	12.0
REINF. EFFICIENCY						
γ	1.63	1.40	1.71	1.48	1.71	1.58
ALLOW. STRESS[1]						
F_g, ksi	17.8	68.3	18.2	63.1	18.2	61.6
MARGIN OF SAFETY	+1.22	+0.24	+0.82	+0.05	+0.82	+0.03

NOTES:

(1) ALLOWABLE FUSELAGE STRESS $(F_g) = \gamma (n\, k_0)/\sqrt{L}$
WHERE: n = 1, FOR CIRCUMFERENTIAL CRACKS
n = 1/2, FOR LONGITUDINAL CRACKS

(2) DESIGN LOAD CONDITION: START-OF-CRUISE

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A summary of the results of the fuselage fail-safe analysis conducted for the Task I initial screening is shown in Table 13-25. Positive margins are indicated; hence, there is no weight penalty associated with these designs.

The initial screening of the fuselage concepts resulted in the selection of a structural arrangement composed of the zee-stiffened and closed hat-stiffened panel concepts. With the zee-stiffened concept employed for the fuselage forebody region and the hat-stiffened design for the midbody and aftbody regions. The selection of this arrangement was based on the results of strength analysis since no weight penalties were associated with the fail-safe or sonic fatigue analyses.

The most promising fuselage arrangement surviving the initial screening was analyzed in greater depth during the next Task I phase (Detailed Concept Analysis). The strength analysis (Section 12) defined the internal forces/stresses and the required panel dimensions. These dimensions are shown in Table 13-26 for the four point design regions. A summary of the stress state at various circumferential locations is shown in Table 13-27.

The fail-safe analysis conducted on the hat-stiffened concept at FS 2500 is shown in Table 13-28. Positive margins are indicated except for the side panel with a circumferential crack. A negative margin of 55-percent is noted at this location, the weight penalty associated with this location is shown on the following summary table. Table 13-29 summarizes the results of the fail-safe analysis conducted during the Task I Detail Concept Analysis. Negative margins are indicated at each point design region, with the maximum valve occurring at FS 2500 (negative 55-percent). These areas required additional structure to meet the fail-safe requirement, the weight penalty associated with this structure is also shown on this table. The weight penalty associated with the highest negative margin area (side panel at FS 2500) is 1.43 lb/sq ft.

TABLE 13-25. SUMMARY OF FUSELAGE PANEL FAIL-SAFE ANALYSES, INITIAL SCREENING

DESIGN CONCEPT	PANEL LOCATION	POINT DESIGN REGION	TYPE OF CRACK	CRACK LENGTH, (IN.)	(1)(2) $\Sigma A_e/t$ (IN.)	REINFORCEMENT EFFICIENCY γ	MARGIN OF SAFETY	WEIGHT PENALTY (LB/SQ. FT)
ZEE-STIFF. CONCEPT	TOP ℄ PANEL (MAX. TENSION)	FS 2000 AND 3000	LONG. CIRCUM.	40.0 8.0	1.44 2.14	1.63 1.37	+1.22 +0.38	NONE NONE
		FS 2500	LONG. CIRCUM.	40.0 8.0	1.44 2.60	1.63 1.40	+1.22 +0.24	NONE NONE
HAT-STIFF. (OPEN) CONCEPT		FS 2000 AND 3000	LONG. CIRCUM.	40.0 10.0	1.80 2.85	1.71 1.47	+0.82 +0.22	NONE NONE
		FS 2500	LONG. CIRCUM.	40.0 10.0	1.80 3.20	1.71 1.48	+0.82 +0.05	NONE NONE
HAT-STIFF. (CLOSED) CONCEPT		FS 2000 AND 3000	LONG. CIRCUM.	40.0 12.0	1.80 3.35	1.71 1.53	+0.82 +0.10	NONE NONE
		FS 2500	LONG. CIRCUM.	40.0 12.0	1.80 4.25	1.71 1.58	+0.82 +0.03	NONE NONE

(1) ΣA_e = SUM OF THE EFFECTIVE AREAS
(2) t = SKIN THICKNESS

TABLE 13-26. FUSELAGE PANEL GEOMETRY - DETAILED CONCEPT ANALYSIS

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	FUSELAGE PANEL DIMENSION						
			b_s (IN.)	t_s (IN.)	C (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	t (IN.)
FS 750	ZEE-STIFFENED	TOP SIDE BOTTOM	4.0 4.0 4.0	.036 .036 .036	.55 .55 .55	.75 .75 .75	1.00 1.00 1.00	.036 .036 .036	.056 .056 .056
FS 2000	HAT-STIFFENED	TOP SIDE	6.0 6.0	.080 .063	1.5 1.5	.80 .75	1.25 1.25	.070 .040	.145 .099
FS 2500	HAT-STIFFENED	TOP SIDE	6.0 6.0	.100 .063	1.5 1.5	.80 .75	1.25 1.25	.090 .050	.184 .109
FS 3000	HAT-STIFFENED	TOP SIDE BOTTOM	6.0 6.0 6.0	.080 .063 .090	1.5 1.5 1.5	.80 .75 .90	1.25 1.25 1.25	.070 .040 .090	.145 .099 .177

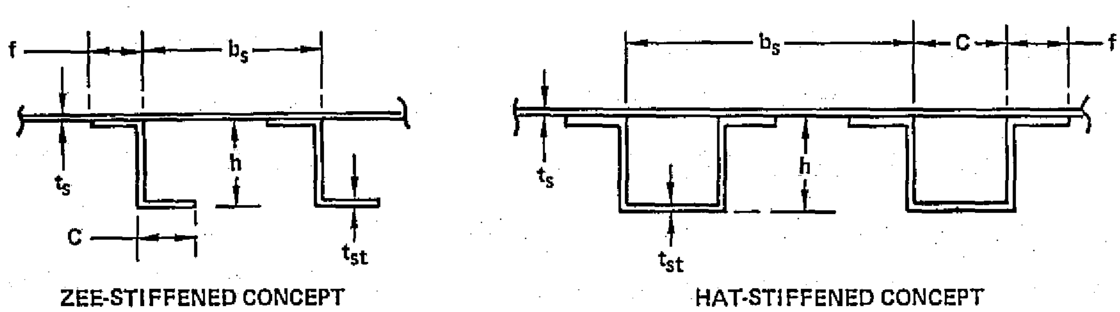


TABLE 13-27. SUMMARY OF FUSELAGE PANEL STRESSES, DETAILED CONCEPT ANALYSIS

LOCATION	FUSELAGE SKIN STRESSES (ULT) – KSI(1)											
	FS 750			FS 2000			FS 2500			FS 3000		
	(2) f_x	(3) f_θ	(2) f_{xy}	(2) f_x	(3) f_θ	(2) f_{xy}	(2) f_x	(3) f_θ	(2) f_{xy}	(2) f_x	(3) f_θ	(2) f_{xy}
TOP	31.8	24.6	1.6	80.2	12.5	5.2	90.4	12.4	7.0	80.2	12.4	5.2
SIDE				12.4	16.7	21.6	11.3	15.9	32.1	12.4	15.1	21.6
BOTTOM				–	–	–	–	–	–	-65.9	11.2	4.6

(1) LIMIT STRESS = 2/3 ULTIMATE STRESS

(2) MAXIMUM AXIAL STRESS (f_x) AND SHEAR STRESS (f_{xy}) CORRESPOND TO START-OF-CRUISE CONDITION

(3) MAXIMUM HOOP STRESS (f_θ) CORRESPONDS TO M1.20 DESCENT CONDITION

TABLE 13-28. FUSELAGE PANEL FAIL-SAFE ANALYSIS, DETAILED CONCEPT ANALYSIS

ITEM	POINT DESIGN REGION – FS 2500			
CRACK TYPE	LONGITUDINAL		CIRCUMFERENTIAL	
LOCATION	TOP	SIDE	TOP	SIDE
PANEL GEOMETRY				
t_s, in.	.090	.063	.090	.063
b_s, in.	6.00	6.00	6.00	6.00
REINFORCEMENT AREA (Ae), in^2.	.036	.035	.175	.090
LIMIT STRESSES				
f_x, ksi.	–	–	60.3	7.5
f_θ, ksi.	8.3	10.6	–	–
f_{xy}, ksi.	–	–	4.7	21.4
FRACTURE TOUGHNESS				
k_o, ksi – $\sqrt{in.}$	136	128	136	128
1/2 k_o, ksi – $\sqrt{in.}$	68	64	–	–
CRACK LENGTH, in.	40.0	40.0	12.0	12.0
REINFORCEMENT EFFICIENCIES				
γ	1.52	1.58	1.58	1.52
ψ	–	–	1.00	0.34
ALLOWABLE STRESSES				
F_g, ksi.	16.3	16.0	61.8	19.1
F_s, ksi	–	–	30.9	9.5
MARGIN OF SAFETY	+0.97	+0.51	+0.02	-0.55

TABLE 13-29. SUMMARY OF FUSELAGE PANEL FAIL-SAFE ANALYSES, DETAILED CONCEPT ANALYSIS

DESIGN CONCEPT	POINT DESIGN REGION	PANEL LOCATION	TYPE OF CRACK	CRACK LENGTH, (IN.)	$\Sigma Ae/t$ (1)(2) (IN.)	REINFORCEMENT EFFICIENCIES		MARGIN OF SAFETY	WEIGHT(3) PENALTY (LB/SQ FT)
						γ	ψ		
ZEE-STIFF. CONCEPT	FS 750	TOP	LONG.	40.0	1.33	1.63	–	-0.12	0.16
		TOP	CIRCUM.	12.0	2.00	1.42	1.00	+1.10	NONE
HAT-STIFF. CONCEPT	FS 2000	TOP	LONG.	40.0	0.98	1.55	–	+0.97	NONE
		SIDE	LONG.	40.0	1.14	1.58	–	+0.53	NONE
		TOP	CIRCUM.	12.0	3.38	1.54	1.00	+0.11	NONE
		SIDE	CIRCUM.	12.0	2.28	1.45	0.49	-0.09	0.16
HAT-STIFF. CONCEPT	FS 2500	TOP	LONG.	40.0	0.80	1.52	–	+0.97	NONE
		SIDE	LONG.	40.0	1.11	1.58	–	+0.51	NONE
		TOP	CIRCUM.	12.0	3.89	1.58	1.00	+0.02	NONE
		SIDE	CIRCUM.	12.0	2.86	1.52	0.34	-0.55	1.43
HAT-STIFF. CONCEPT	FS 3000	TOP	LONG.	40.0	0.98	1.55	–	+0.99	NONE
		SIDE	LONG.	40.0	1.08	1.57	–	+0.57	NONE
		BOTTOM	LONG.	40.0	0.91	1.54	–	+1.23	NONE
		TOP	CIRCUM.	12.0	3.38	1.54	1.00	+0.11	NONE
		SIDE	CIRCUM.	12.0	2.79	1.51	0.49	-0.05	0.09
		BOTTOM	CIRCUM.	12.0	–	–	–	+ HIGH	NONE

NOTES:

(1) ΣAe = SUM OF EFFECTIVE AREAS

(2) t = SKIN THICKNESS

(3) WEIGHT PENALTY (ΔW) = 23.04 $\Delta\bar{t}$, FOR 6AL-4V TITANIUM; UNITS - LB/SQ FT
$\Delta\bar{t} = t_{FS} - t_{STR}$; UNITS – IN.

Fail-Safe Analysis - Task II

The Task IIB results of the fail-safe analysis on the strength/stiffness airplane are presented in the following sections. Fail-Safe analyses were not conducted during the Task IIA configuration change investigation and only a cursory analysis, indicative of that stage of design, was conducted on the strength design airplane of Task IIB.

The structural approach incorporated on the Task IIB airplane was a hybridization of the Task I Chordwise and Monocoque wing designs utilizing both metallic and composite materials with the fuselage being conventional skin/stringer design. This hybrid arrangement consists of the following concepts:

- Wing-forward and aft boxes: metallic chordwise stiffened wing panels, convex-beaded concept, with submerged titanium and titanium/composite reinforced spar caps.
- Wing tip: Monocoque wing panels, 6Al-4V titanium honeycomb sandwich panels with aluminum brazed core, with metallic substructure and embedded rib/spar caps.
- Fuselage-shell: Conventional skin/stringer/frame design utilizing Ti-6Al-4V material.

Fail-safe analyses were conducted on these above concepts at the six wing and four fuselage point design regions. The wing locations are identical to the Task I locations shown in Figure 13-13; whereas, the fuselage locations were altered and are presented in Figure 13-19.

The method of analysis and the available data used in determining the residual strength is as outlined in Figure 13-10.

Wing Panel Analysis - Each of the surface panel and spar cap concepts associated with the point design regions were analyzed on the final design airplane (strength/stiffness). The convex-beaded concept was employed at point design regions 40322, 40236 and 40536; with regions 41036, 41316, and 41348 being honeycomb sandwich panels.

In addition to a summary of the results, the basic assumptions, geometry, stress levels, and sample analyses are included for the point design regions.

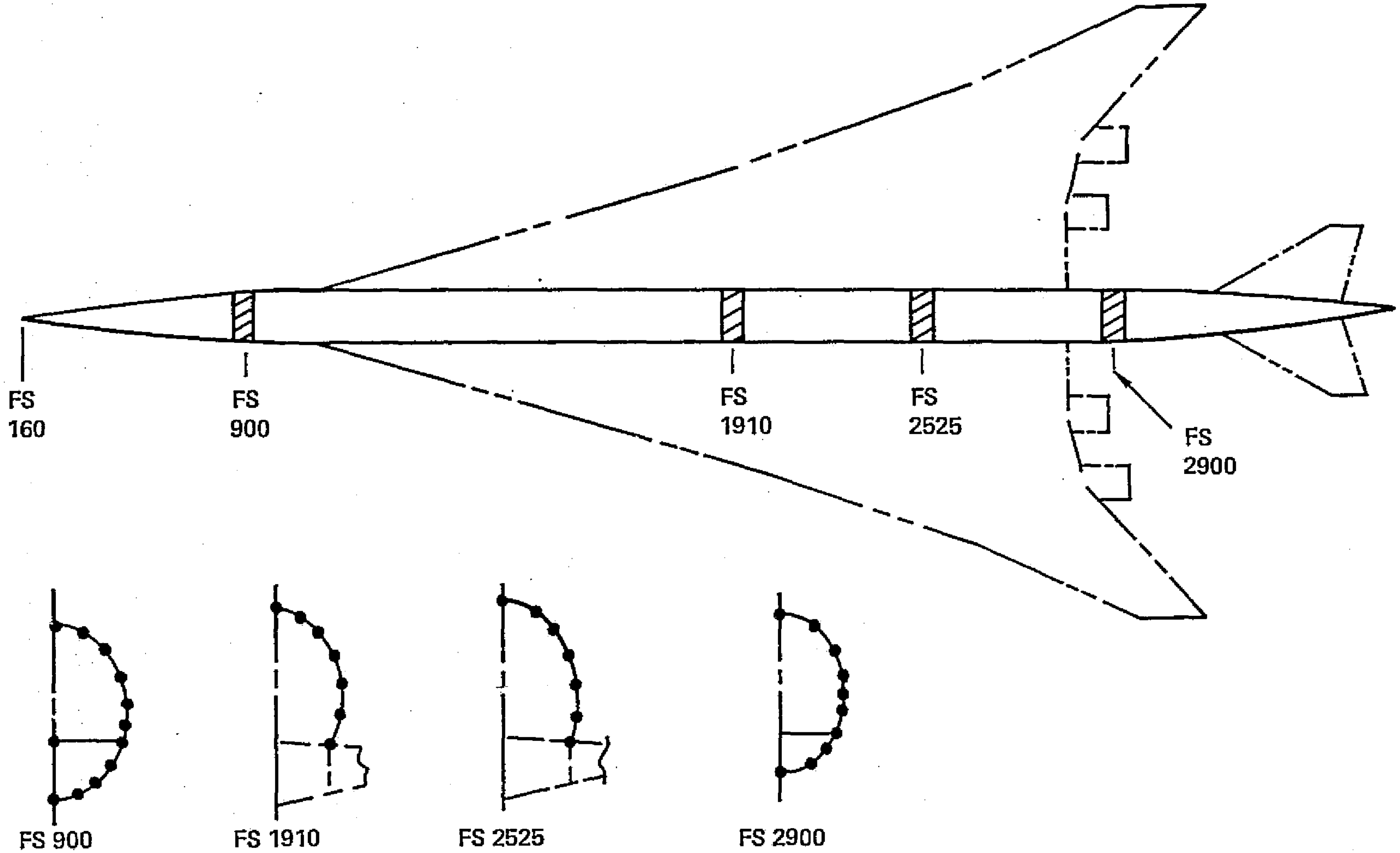


Figure 13-19. Definition of Fuselage Point Design Regions - Task IIB

The panel geometry for convex-beaded and honeycomb sandwich panels are shown in Tables 13-30 and 13-31. These data reflect the results of the strength and stiffness requirements imposed on the Final Design airplane. All designs are strength designs with the exception of the honeycomb sandwich panels at regions 41316 and 41348 which are the stiffness designs resulting from the flutter optimization investigation.

Preliminary fail-safe analyses of these designs indicated several panels were deficient in meeting the fail-safe criteria. For these deficient regions, 40536 and 41036, the panel geometry and associated stress level were adjusted with the new panel geometry being shown in Table 13-32. The weight penalties associated with these geometry changes and any added weight penalties associated with fail-safe requirements are included in the final results. The flight conditions and stress levels for the maximum tension condition are shown in Table 13-33. These skin stresses are ultimate values and, after reduction to limit values, are used as the basis for the fail-safe analysis. With reference to this table, Conditions 12 and 14 (M1.25 Climb condition at 2.5-g and -1-g) are the predominant tension conditions for the lower and upper surface panels respectively. The exception being the maximum tension conditions for the upper and lower panels at point design region 40236 and the lower panel at region 40322. The critical tension conditions for these regions are the M 1.25 climb condition at V_C and the start-of-cruise condition respectively.

For the convex-beaded surface panels, the outer skin was treated as a flat panel with the inner beaded skin considered to be the reinforcement. The effective area (A_e) of the reinforcement was taken to be one-third of inner-bead with the reinforcement parameter ($\Sigma A_e/t$) equal to two-times the effective area divided by the outer skin thickness.

A damaged condition of a three-pitch outer skin crack with two broken reinforcing stiffeners (inner beads) was selected. This resulted in crack lengths between 7-inches to 10-inches for the convex-beaded panels.

The outer skin fracture toughness value (k_o), and the reinforcement efficiency (γ) and shear correction factor (ψ) were determined as outlined in Figure 13-10.

TABLE 13-30. WING PANEL GEOMETRY - TASK IIB
CHORDWISE, CONVEX-BEADED PANELS

DESIGN DATA	POINT DESIGN REGIONS					
	40322		40236		40536	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
RIB	60.0	60.0	60.0	60.0	60.0	60.0
SPAR	22.7	22.7	21.2	21.2	21.2	21.2
DIMENSIONS						
t_L, in.	.013	.015	.015	.020	.023	.019
t_u, in.	.015	.020	.015	.020	.026	.020
R_L, in.	.80	1.00	.80	1.00	.90	.70
θ, degrees	87	87	87	87	87	87
b, in.	.75	.75	.75	.75	.75	.75
pitch, in.	2.35	2.75	2.35	2.75	2.55	2.15
WEIGHT DATA						
$\bar{t}$, in.	.033	.041	.036	.048	.058	.046
W, lb./sq.ft.	.760	.945	.829	1.11	1.34	1.05
CRITICAL DESIGN COND.	12	20	16	16	12	12

DIMENSIONS:

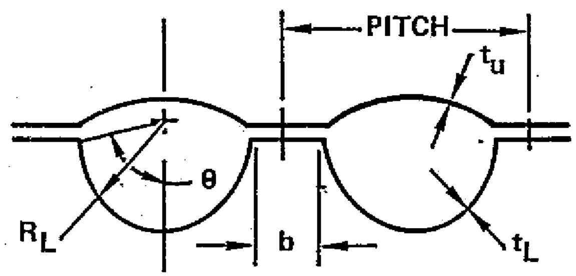


TABLE 13-31. WING PANEL GEOMETRY - TASK IIB
HONEYCOMB SANDWICH PANELS

DESIGN DATA	POINT DESIGN REGIONS					
	41036		41316		41348	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
RIB	60.0	60.0	40.0	40.0	40.0	40.0
SPAR	21.2	21.2	40.0	40.0	30.0	30.0
DIMENSIONS						
H, in.	.642	.202	1.00	.500	1.00	.500
t_1, in.	.026	.023	.062	.075	.068	.068
t_2, in.	.018	.028	.062	.075	.068	.068
t_c, in.	.002	.002	.002	.002	.002	.002
S, in.	.275	.500	.500	.500	.500	.500
WEIGHT DATA						
$\bar{t}$, in.	.052	.052	.131	.153	.143	.139
W, lb./sq.ft.	1.20	1.20	3.02	3.52	3.29	3.20
CRITICAL DESIGN COND.	12	12	FLUTTER	FLUTTER	FLUTTER	FLUTTER

DIMENSIONS

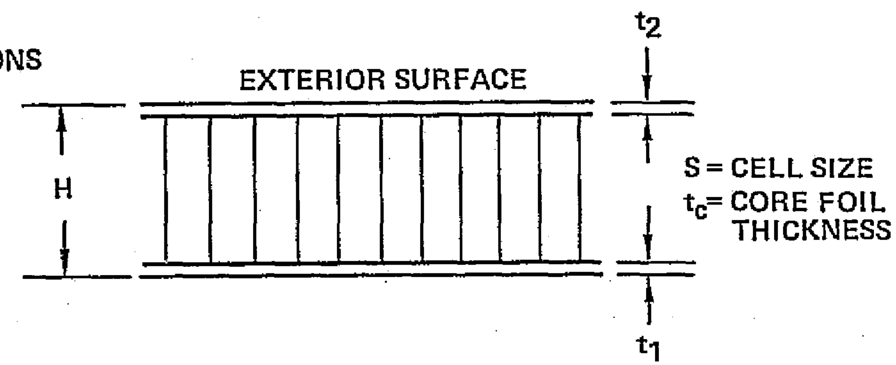
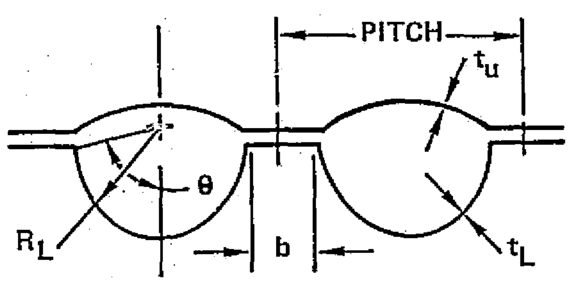
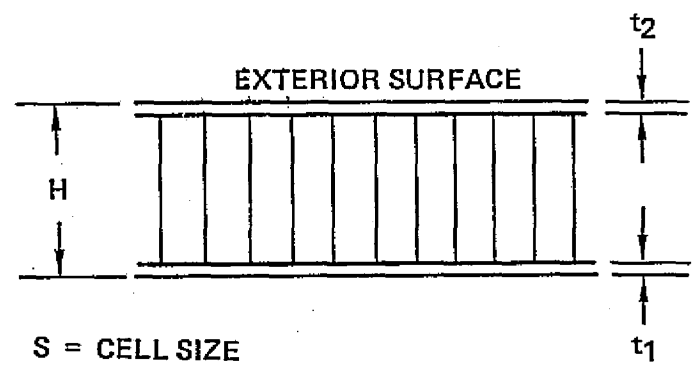


TABLE 13-32. REVISED WING PANEL GEOMETRY - TASK IIB

DESIGN DATA	CONVEX BEADED 40536 UPPER	CONVEX BEADED 40536 LOWER	DESIGN DATA	HONEYCOMB SANDWICH 41036 UPPER	HONEYCOMB SANDWICH 41036 LOWER
SPACING, in.			SPACING, in.		
RIB	60.0	60.0	RIB	60.0	60.0
SPAR	21.2	21.2	SPAR	21.2	21.2
DIMENSIONS			DIMENSIONS		
t_L, in.	.025	.035	H, in	1.00	0.50
t_u, in.	.031	.060	t_1, in.	.033	.041
R_L, in.	.80	1.25	t_2, in.	.033	.041
θ, deg.	87	87	t_c, in.	.002	.002
b, in.	.75	.75	S, in.	.500	.500
pitch, in.	2.35	3.25			
WEIGHT DATA			WEIGHT DATA		
$\bar{t}$, in.	.065	.110	$\bar{t}$, in.	.073	.085
W, lb./sq.ft.	1.50	2.53	W, lb./sq.ft.	1.68	1.96
DESIGN COND.	FAIL-SAFE	FAIL-SAFE	DESIGN COND.	FAIL-SAFE	FAIL-SAFE





S = CELL SIZE
t_c = CORE FOIL THICKNESS

TABLE 13-33. SUMMARY OF WING PANEL SKIN STRESSES - TASK IIB

PANEL CONCEPT	POINT DESIGN REGIONS	SPACING (in.)		PANEL SKIN STRESSES (ULT.) — ksi. (1)							
				UPPER SURFACE				LOWER SURFACE			
		SPAR	RIB	COND. (2)	f_x	f_y	f_{xy}	COND. (2)	f_x	f_y	f_{xy}
CHORDWISE CONVEX-BEADED	40322	22.7	60.0	14	3.15	–	1.20	20	24.20	–	8.27
	40236	21.2	60.0	16	14.58	–	9.55	16	52.95	–	18.42
	40536	21.2	60.0	14	18.98	–	22.95	12	10.00	–	18.60
MONOCOQUE HONEYCOMB SANDWICH	41036	21.2	60.0	14	19.90	35.40	20.90	12	16.40	41.40	23.30
	41316	40.0	40.0	14	7.94	52.20	16.65	12	9.30	72.70	19.80
	41348	30.0	40.0	14	5.44	26.53	9.50	12	11.30	48.40	16.80

1. LIMIT STRESS = 2/3 ULTIMATE STRESS
2. CRITICAL TENSION FLIGHT CONDITIONS, NASTRAN CONDITION NUMBERS

The detailed fail-safe calculations for the convex-beaded panels are presented in Table 13-34 and indicated positive margins for all regions. A minimum positive margin of 3-percent is noted on the upper surface panel at region 40536. The panel geometry analyzed for region 40536 reflects the adjusted panel cross-sections shown in Table 13-30. The weight penalty associated with this change is included in the summary table of the wing panel results.

The honeycomb sandwich panels at point design regions 41036, 41316, and 41348 were analyzed on the Final Design airplane. The panel geometry and dimensions were previously shown in Tables 13-31 and 13-32. These panels incorporated the densified core design for attachment to rib and spar webs. Figure 13-20 presents the minimum design requirements for these attachment areas.

The damaged condition was a two-bay crack with a broken reinforcement (strap or spar/rib attachment) and both face sheets damaged. A maximum crack length of 20 inches was assumed.

The Task I monocoque panel damage configurations shown in Figure 13-16 are appropriate for the Task II designs.

The effective area (A_e) of the strap or densified core attachment, see Figure 13-20, was considered to be the area of the straps/doublers. For the densified core design the core was considered to be ineffective.

For the three honeycomb panel regions the face sheet thicknesses were equal and the sum of these thicknesses ($t_1 + t_2$) was taken to be the skin thickness (t) in calculating the ratio $\Sigma A_e/t$. The fracture toughness values, and the parameters γ and ψ were determined from Figure 13-10.

Table 13-35 presents the fail-safe calculation for the honeycomb sandwich panels. The panel geometry specified on this table for region 41036 reflects the adjusted panel cross-sections defined in Table 13-32. A minimum positive margin of safety of 1-percent is shown for the lower surface panels at point design regions 41036 and 41316, all other margins range between 5-percent to 50-percent. For this analysis the required strap area has been determined to obtain a positive margin with a minimum strap area of 0.06 inches being defined.

TABLE 13-34. WING PANEL FAIL-SAFE ANALYSIS - TASK IIB
CONVEX-BEADED PANELS

ITEM	POINT DESIGN REGIONS					
	40322		40236		40536	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
SPAR	22.7	22.7	21.2	21.2	21.2	21.2
RIB	60.0	60.0	60.0	60.0	60.0	60.0
DISTANCE BETWEEN UNBROKEN BEADS, in.	7.05	8.25	7.05	8.25	7.05	9.75
CRACK LENGTH (L), in.	7.05	8.25	7.05	8.25	7.05	9.75
$\sqrt{L}$, $\sqrt{in.}$	2.66	2.87	2.66	2.87	2.66	3.12
LIMIT STRESSES, ksi						
COND. NO.	14	20	16	16	14	12
f_x	2.10	16.13	9.72	35.30	12.66	6.70
f_{xy}	0.80	5.51	6.37	12.28	15.30	12.40
f_{xy}/f_x	0.38	0.34	0.66	0.35	1.21	1.85
EFFECTIVE AREA (Ae), in^2	.014	.019	.016	.025	.026	.053
SKIN THICKNESS, in.	.015	.020	.015	.020	.031	.060
$\Sigma Ae/t$, in.	1.87	1.90	2.13	2.50	1.68	1.77
FRACTURE TOUGHNESS k_o, ksi - $\sqrt{in.}$	80	84	80	84	104	127
REINFORCEMENT EFF.						
γ	1.33	1.35	1.35	1.40	1.32	1.37
ψ	0.91	0.93	0.79	0.93	0.61	0.47
ALLOWABLE STRESSES, ksi						
$F_g = \psi \gamma k_o/\sqrt{L}$	36.4	36.7	32.1	38.1	31.5	26.2
$F_s = 1/2 F_g$	18.2	18.4	16.1	19.1	15.8	13.1
MARGIN OF SAFETY	HIGH	+1.28	+1.52	+0.08	+0.03	+0.06

<table>
<tr><th>REINFORCEMENT DESIGN</th><th>REINFORCEMENT AREA (A_R), in^2</th><th>Ae in^2</th></tr>
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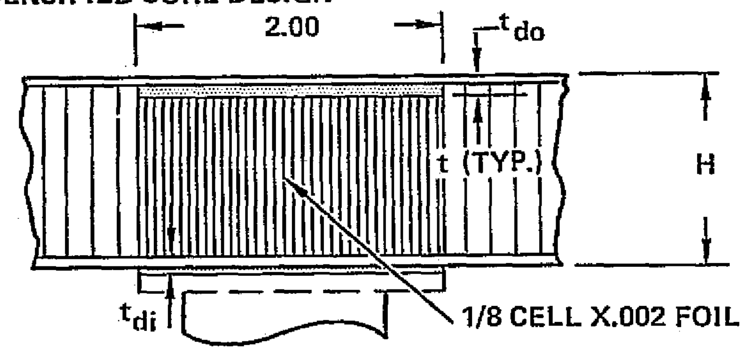
</td>
<td>EQUATION: $A_R = 2.00\,(t_{do} + t_{di})$
<table>
<tr><th>REGION</th><th>SURF.</th><th>t_{do}</th><th>t_{di}</th><th>A_R</th></tr>
<tr><td>41036</td><td>UPPER
LOWER</td><td>.032
.022</td><td>.015
.015</td><td>.094
.074</td></tr>
<tr><td>41316</td><td>UPPER
LOWER</td><td>.023
.024</td><td>.020
.020</td><td>.086
.088</td></tr>
<tr><td>41348</td><td>UPPER
LOWER</td><td>.023
.023</td><td>.020
.020</td><td>.086
.086</td></tr>
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<td>$A_e = A_R$</td>
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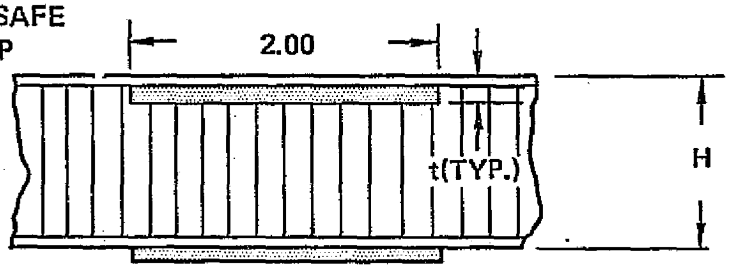
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CORE NEGLECTED</td>
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Figure 13-20. Honeycomb Sandwich Insert Geometry and Data - Task IIB

TABLE 13-35. WING PANEL FAIL-SAFE ANALYSIS - TASK IIB
HONEYCOMB SANDWICH PANELS

ITEM	POINT DESIGN REGIONS					
	41036		41316		41348	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
SPAR	21.2	21.2	40.0	40.0	30.0	30.0
RIB	60.0	60.0	40.0	40.0	40.0	40.0
PANEL DIMENSIONS						
t_2, in.	.033	.041	.062	.075	.068	.068
t_1, in.	.033	.041	.062	.075	.068	.068
REINFORCEMENT AREA						
SPAR CAP, in^2	.094	.074	.086	.088	.086	.086
STRAP, in^2	.060	.070	.060	.330	—	.060
EFFECTIVE AREA						
Ae, in^2	.06	.07	.06	.33	0.0	0.06
$\Sigma Ae/(t_1 + t_2)$, in.	1.82	1.71	1.00	4.4	0.0	0.88
LIMIT STRESSES, ksi						
COND. NO.	14	12	14	12	14	12
f_x	13.3	10.93	5.29	6.20	3.63	7.50
f_y	23.6	27.60	34.80	48.47	17.70	32.30
f_{xy}	13.9	15.50	11.10	13.20	6.33	11.20
f_{xy}/f_y	0.59	0.56	0.32	0.27	0.36	0.35
FRACTURE TOUGHNESS						
$k_o(t_{MIN})$, ksi - $\sqrt{in.}$	106	114	128	132	130	130
CRACK GEOMETRY						
DIRECTION	CHORD	CHORD	CHORD	CHORD	CHORD	CHORD
LENGTH (L), in.	20.0	20.0	20.0	20.0	20.0	20.0
$\sqrt{L}$, $\sqrt{in.}$	4.47	4.47	4.47	4.47	4.47	4.47
REINFORCEMENT EFF.						
γ	1.52	1.48	1.37	1.74	1.00	1.36
ψ	0.81	0.83	0.93	0.95	0.91	0.91
ALLOWABLE STRESSES, ksi						
$F_g = \psi \gamma K_o/\sqrt{L}$	29.2	31.3	36.5	48.8	26.5	36.0
$F_s = 1/2 F_g$	14.6	15.6	18.2	24.4	13.2	18.0
MARGIN OF SAFETY	+0.05	+0.01	+0.05	+0.01	+0.50	+0.11

The calculations of the weight penalties associated with the wing panel concepts are shown in Table 13-36. This includes both the convex-beaded and honeycomb sandwich designs. This table includes the minimum design requirements (strength) and the fail-safe requirements for the spar caps, straps, and panels. All honeycomb panels require fail-safe straps at approximately 10-inch spacing. The exception being the lower surface panel at point design region 41348 which is capable of support limit load with a 20 inch crack and requires no additional reinforcement. The convex-beaded panel concepts require no additional reinforcement other than the panel geometry revisions for the upper and lower panels at region 40536.

A summary of the wing panel fail-safe results is presented in Table 13-37. This table summarizes the pertinent fail-safe data, margins of safety, and the corresponding weight penalties. The largest weight penalty associated with the convex-beaded concept is 1.47 pounds/square foot for the lower panel at region 40536. Similarly, the maximum penalty for the honeycomb panel concept is 0.84 pounds/square foot for the lower panel at region 41036. No added structural reinforcement (weight penalty) is required on the convex-beaded concept at regions 40322 and 40236 or the lower surface honeycomb sandwich panel at region 41348.

Wing Spar Analysis - The composite reinforced spar cap in the aft wing box were analyzed to define their damage tolerance. This included the spar caps associated with point design regions 40236, 40536, and 41036.

The detail dimensions of these titanium caps reinforced with unidirectional Boron/polyimide (B/PI) are shown in Table 13-38. These sections incorporated a constant metal substrate with the area of the B/PI reinforcement varied to meet the strength requirements. A maximum composite thickness of approximately 0.60 inch was required for the lower surface spars at regions 40236 and 40536. Conversely, the minimum thicknesses occur on outboard region at 41036.

Similar to the analysis conducted in Task I, a simplified strength analysis utilizing the multiple element characteristics of these spars was conducted to define the damage tolerance trends. Table 13-39 summarizes the results of this analysis. The flight condition and corresponding loads for the maximum tension case were defined by scanning the final design loads, see Section 11, Point Design Environment. As in the wing panel analysis, Conditions 12 and 14 were the most critical flight conditions.

TABLE 13-36. WING PANEL WEIGHT PENALTY — TASK IIB

POINT DESIGN REGION	DESIGN	SURFACE	PANEL DIMENSIONS		MIN. DESIGN REQUIREMENTS		FAIL-SAFE REQUIREMENTS				WEIGHT PENALTY (lb./sq. ft.)
			SPAR SPACING b, (in.)	RIB SPACING a, (in.)	PANEL THK., $\bar{t}$ (in.)	SPAR AREA, (in^2)	PANEL THK. (in.)	SPAR AREA, (in^2)	STRAP AREA, (in^2)	NO. STRAPS	
40322	CONVEX-BEADED	UPPER	22.7	60.0	0.033	—	—	—	—	—	NONE
		LOWER	22.7	60.0	0.041	—	—	—	—	—	NONE
40236	CONVEX-BEADED	UPPER	21.2	60.0	0.036	—	—	—	—	—	NONE
		LOWER	21.2	60.0	0.048	—	—	—	—	—	NONE
40536	CONVEX-BEADED	UPPER	21.2	60.0	0.058	—	0.065	—	—	—	0.16
		LOWER	21.2	60.0	0.046	—	0.110	—	—	—	1.47
41036	HONEYCOMB SANDWICH	UPPER	21.2	60.0	0.052	0.094	0.073	0.060	0.060	1	0.55
		LOWER	21.2	60.0	0.052	0.074	0.085	0.070	0.070	1	0.84
41316	HONEYCOMB SANDWICH	UPPER	40.0	40.0	0.131	0.086	—	0.060	0.060	3	0.10
		LOWER	40.0	40.0	0.153	0.088	—	0.330	0.330	3	0.71
41348	HONEYCOMB SANDWICH	UPPER	30.0	40.0	0.143	0.086	—	—	—	2	NONE
		LOWER	30.0	40.0	0.139	0.086	—	0.060	0.060	2	0.09

WEIGHT PENALTY EQUATION (EQUIVALENT PANEL WEIGHT)

$$\Delta W = 144\rho\left[\left(\bar{t}_{PANEL,\,FS} - \bar{t}_{PANEL,\,MIN}\right) + \left(A_{SPAR,\,FS} - A_{SPAR,\,MIN} + n\,A_{STRAP}\right)/b\right]$$

WHERE:

ρ = MATERIAL DENSITY, 0.160 lb./in^3

$\bar{t}_{PANEL,\,FS}$ = PANEL THICKNESS REQUIRED FOR FAIL-SAFE

$\bar{t}_{PANEL,\,MIN}$ = PANEL THICKNESS, MINIMUM DESIGN REQUIREMENTS

b = SPAR SPACING

$A_{SPAR,\,FS}$ = SPAR CAP AREA REQUIRED FOR FAIL-SAFE

$A_{SPAR,\,MIN}$ = SPAR CAP AREA, MINIMUM DESIGN REQUIREMENT

A_{STRAP} = STRAP AREA REQUIRED FOR FAIL-SAFE

n = NUMBER STRAPS

TABLE 13-37. SUMMARY OF WING PANEL FAIL-SAFE ANALYSIS HYBRID ARRANGEMENT — TASK IIB

DESIGN CONCEPT	POINT DESIGN REGION	WING SURFACE	SPACING SPAR (in.)	SPACING RIB (in.)	CRACK LENGTH (in.)	$\Sigma Ae/t$ (in.)	REINFORCEMENT EFFICIENCY γ	REINFORCEMENT EFFICIENCY ψ	MARGIN OF SAFETY	WEIGHT PENALTY (lb./sq. ft.)
CONVEX-BEADED PANELS	40322	UPPER	22.7	60.0	7.05	1.87	1.33	0.91	LARGE	NONE
		LOWER	22.7	60.0	8.25	1.90	1.35	0.93	+1.28	NONE
	40236	UPPER	21.2	60.0	7.05	2.13	1.35	0.79	+1.52	NONE
		LOWER	21.2	60.0	8.25	2.50	1.40	0.93	+0.08	NONE
	40536	UPPER	21.2	60.0	7.05	1.68	1.32	0.61	+0.03	0.16
		LOWER	21.2	60.0	9.75	1.77	1.37	0.47	+0.06	1.47
HONEYCOMB-SANDWICH PANELS	41036	UPPER	21.2	60.0	20.0	1.82	1.52	0.81	+0.05	0.55
		LOWER	21.2	60.0	20.0	1.71	1.48	0.83	+0.01	0.84
	41316	UPPER	40.0	40.0	20.0	1.00	1.37	0.93	+0.05	0.10
		LOWER	40.0	40.0	20.0	4.40	1.74	0.95	+0.01	0.71
	41348	UPPER	30.0	40.0	20.0	—	1.00	0.91	+0.50	NONE
		LOWER	30.0	40.0	20.0	0.88	1.36	0.91	+0.11	0.09

TABLE 13-38. SPAR CAP GEOMETRY - TASK IIB
COMPOSITE REINFORCED CONCEPT

POINT DESIGN REGION	SPAR SPACING (in.)	SPAR CAP DIMENSIONS					
		h (in.)	b (in.)	H (in.)	W (in.)	t_1 (in.)	t_2 (in.)
40236							
UPPER	21.2	.38	1.00	1.20	2.50	.12	.12
LOWER	21.2	.62	1.00	1.20	2.50	.12	.12
40536							
UPPER	21.2	.38	1.00	1.20	2.50	.12	.12
LOWER	21.2	.58	1.00	1.20	2.50	.12	.12
41036							
UPPER	21.2	.10	1.00	1.20	2.50	.12	.12
LOWER	21.2	.12	1.00	1.20	2.50	.12	.12

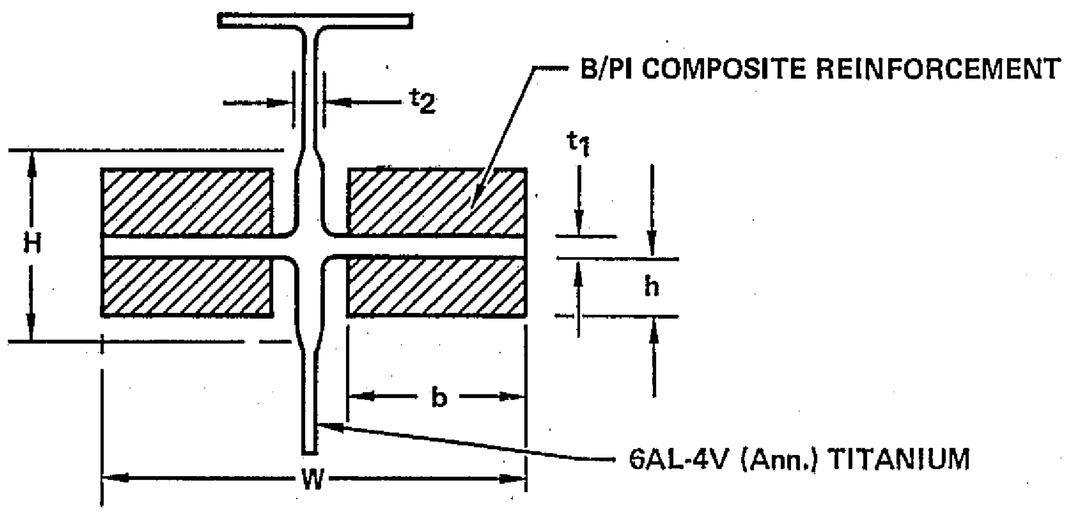


TABLE 13-39 SUMMARY OF SPAR CAP FAIL-SAFE ANALYSES COMPOSITE REINFORCED CONCEPT — TASK IIB

POINT DESIGN REGION	SPAR SPACING (in.)	TOTAL APPLIED(1) (2) LOAD, KIPS			MEMBER ALLOWABLE(3) LOAD, KIPS		DAMAGED CAP(4) ALLOWABLE LOAD (P_A), KIPS		MARGIN(5) OF SAFETY
		COND NO.	P_{ULT}	P_{LIMIT}	METAL ELEMENT P_M	COMPOSITE ELEMENT P_C	BROKEN METAL ELM.	BROKEN COMPOSITE ELM.	
40236									
UPPER	21.2	14	190.2	126.8	35.5	55.9	223.6	203.2	+0.60
LOWER	21.2	12	391.2	260.8	33.6	89.4	357.6	301.8	+0.16
40536									
UPPER	21.2	14	176.3	117.5	35.5	55.5	222.0	202.0	+0.72
LOWER	21.2	12	364.9	243.3	34.5	82.6	330.4	282.3	+0.16
41036									
UPPER	21.2	14	62.3	41.5	37.5	16.0	64.0	85.5	+0.54
LOWER	21.2	12	115.5	77.0	37.1	19.6	78.4	95.9	+0.02

1. MAXIMUM TENSION LOADS
2. LIMIT LOAD = 2/3 ULTIMATE LOAD
3. COMPOSITE ELEMENT LOAD ARE UNIT VALUES, TOTAL LOAD SUSTAINED BY THE COMPOSITE ELEMENTS IS FOUR TIMES THE UNIT VALUES.
4. DAMAGED CAP ALLOWABLES:

   BROKEN METAL MEMBER $P_A = 4 \times P_C$

   BROKEN COMPOSITE MEMBER $P_A = P_M + 3 \times P_C$
5. MARGIN OF SAFETY $= \dfrac{\text{MIN. } P_A}{P_{LIMIT}} - 1$

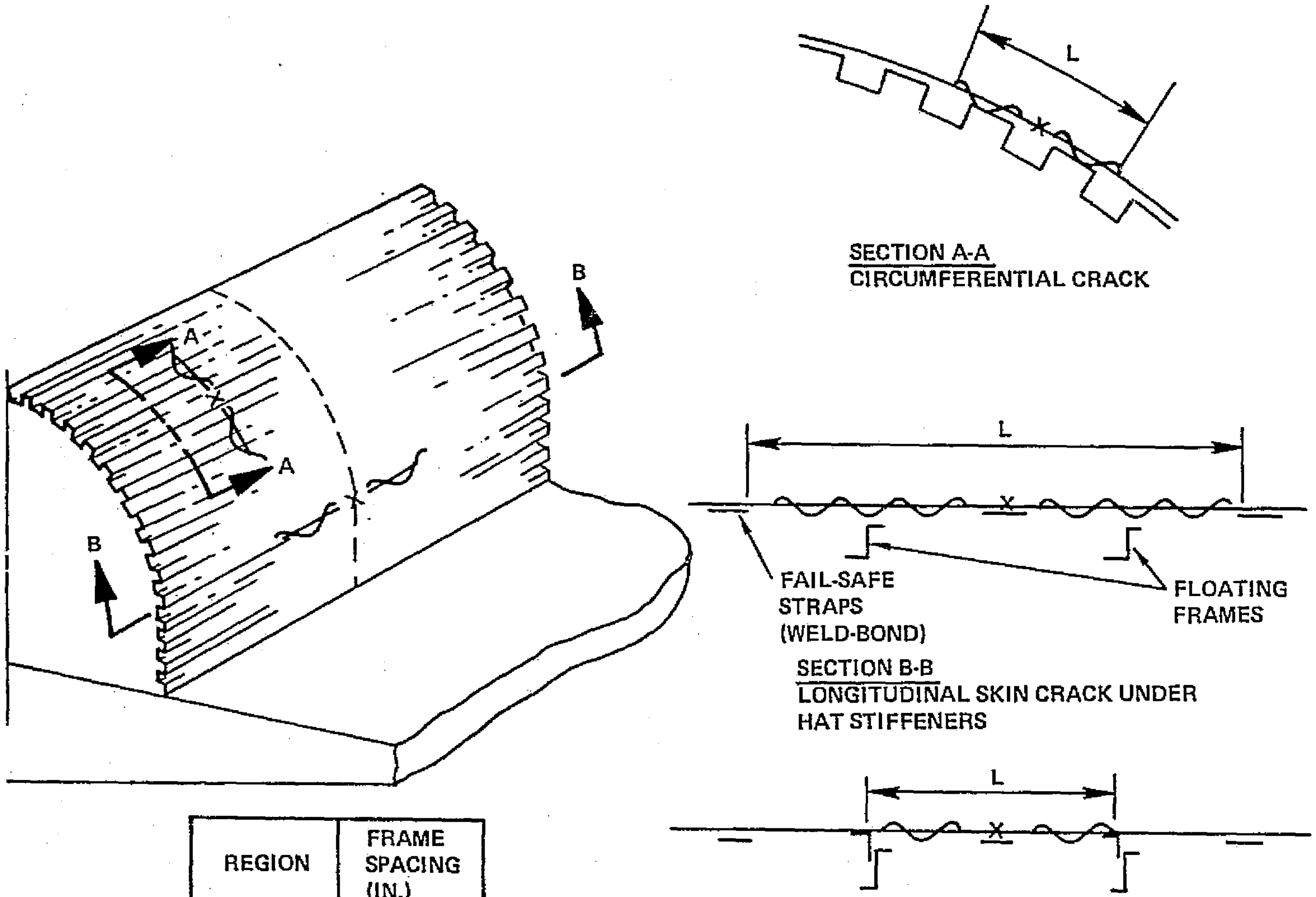
From the strength analysis, the ultimate load carrying capability of each member of the cross-section was defined. For example with reference to Table 13-39, the metallic substrate of the upper spar caps at point design region 40236 has an allowable of 35.5 kips and each of the four composite members are capable of withstanding a load of 55.9 kips.

A damage condition of a single broken member (composite or metal substrate) with the applied limit load redistributed to the remaining undamaged members was considered. The allowable loads for the damaged conditions and margins of safety are also included on Table 13-39. In summary, all composite reinforced caps are fail-safe with a minimum positive margin of 2-percent existing on the lower spar cap at point design region 41036.

Fuselage Analysis - The Task II fuselage fail-safe analyses were conducted to define the damage-tolerance capability of the strength design fuselage and assess the weight penalties associated with meeting the fail-safe requirements.

The analytical method outlined in Figure 13-10 was used with two types of cracks being considered: Circumferential and longitudinal cracks. Similarly to Task I, a 50-percent reduction in fracture toughness (K_O) was imposed on the Task II analyses involving longitudinal cracks, with no reduction in K_O for the circumferential crack conditions.

The assumed fuselage damage configurations are presented in Figure 13-21. For the circumferential crack condition, a damage condition of a two-pitch skin crack with the intermediate stringer broken was considered. This damage condition results in crack lengths from 8 inches to 12 inches. This figure also represents the damage condition for the longitudinal cracks, which is a two-bay crack with the intermediate frame or fail-safe strap broken. As shown on this figure, the most critical damage condition is a longitudinal skin crack under the hat-stiffener. This condition is the most severe since manufacturing difficulties preclude adding straps at the fuselage frame under the hat-stiffeners. Therefore for this condition crack lengths equal to two-full frame spacings, approximately 40 inches, were considered.



REGION	FRAME SPACING (IN.)
FS 900	21.50
FS 1910	22.67
FS 2525	21.25
FS 2900	21.00

Figure 13-21. Fuselage Panel Damage Configuration - Task IIB

Four fuselage point design regions were analyzed in support of the detail engineering studies. The final design fuselage incorporates the zee-stiffened panel concept at FS 750, and the hat-stiffened concept at FS 1910, 2525, and 2900. The panel geometry for each of these stations are shown in Table 13-40. The circumferential location is identified by the equivalent NASTRAN model panel identification number as shown on Figure 13-22.

The fail-safe analysis indicated the locations where basic panel geometry changes were required. As a result, panels 234103 and 234104 at FS 1910 and panels 234805 and 234806 at FS 2525 were amended. The revised section properties for these panels are shown in Table 13-41.

A summary of the maximum tension stresses for selective locations at the point design regions is presented in Table 13-42 and includes the adjusted stresses levels for the revised panels. The critical tension stresses are specified for the crack condition being investigated, i.e., maximum hoop stress for longitudinal cracks and maximum axial (meridional) stress for the circumferential crack condition. For ease in reporting, selective panels at each point design region have been redefined as top, side, and bottom panels. With reference to Figure 13-22, the top definition refers to the upper-most panel at each region, i.e., the panels with the NASTRAN identification numbers (six digit number) ending with the digits 01. Similarly, side and bottom refers to the panels ending with the 06 and 09 digits, respectively. Hereafter, this terminology is used for the fuselage analysis.

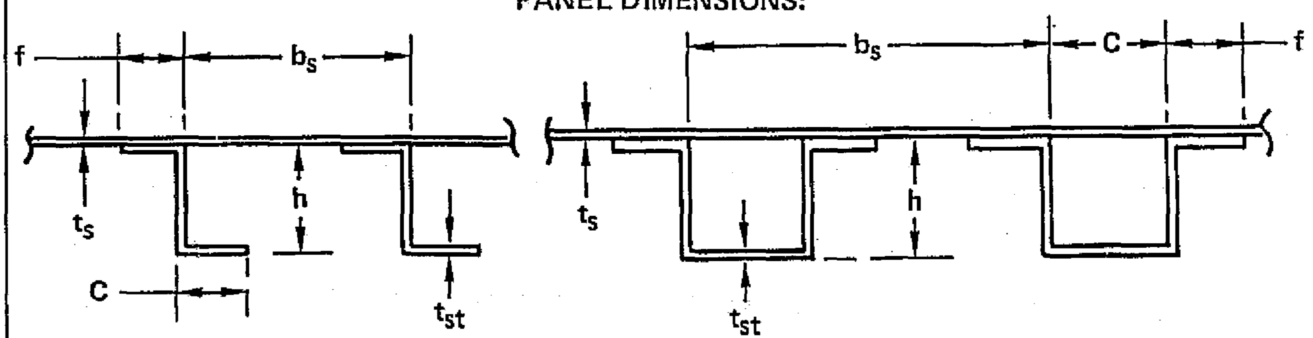
The fuselage was subjected to a comprehensive fail-safe analysis which included all circumferential panels at each of the point design regions. A sample of the fuselage analysis is shown in Table 13-43 for the top and side locations at point design region 2525. This example covers both the longitudinal and circumferential crack conditions. Positive margins are noted for each location, with a minimum margin of 1-percent noted at the side panel for the circumferential crack condition. All other margins are 50-percent or higher. These calculations reflect reinforcement straps for the longitudinal crack analysis and the revised side panel geometry previously discussed and presented in Table 13-41.

A sample of the weight penalty calculations are presented in Table 13-44 for point design region FS 2525. This table indicates the weight parameters (panel thickness and strap area) associated with the individual panel meeting the fail-safe requirements. In addition, the average equivalent panel thickness ($\bar{t}$) and unit weight (w)

TABLE 13-40. FUSELAGE PANEL GEOMETRY - TASK IIB

POINT DESIGN REGION	PANEL CONCEPT	CIRCUMF. LOCATION	FUSELAGE PANEL DIMENSIONS						
			b_s (in.)	t_s (in.)	C (in.)	f (in.)	h (in.)	t_{st} (in.)	$\bar{t}$ (in.)
FS 900	ZEE-STIFFENED	233301-233307	4.0	.036	.55	0.75	1.00	.036	.056
FS 1910	HAT-STIFFENED	234101	6.0	.07	1.5	0.80	1.25	.06	.129
		234102	6.0	.06	1.5	0.80	1.25	.05	.109
		234103	6.0	.04	1.5	0.80	1.25	.04	.079
		234104	6.0	.04	1.5	0.80	1.25	.03	.069
		234105	6.0	.05	1.5	0.80	1.25	.05	.099
		234106	6.0	.06	1.5	0.80	1.25	.06	.119
FS 2525	HAT-STIFFENED	234801	6.0	.07	1.5	0.80	1.25	.08	.149
		234802	6.0	.06	1.5	0.80	1.25	.06	.119
		234803	6.0	.05	1.5	0.80	1.25	.05	.099
		234804	6.0	.04	1.5	0.80	1.25	.03	.069
		234805	6.0	.04	1.5	0.80	1.25	.03	.069
		234806	6.0	.04	1.5	0.80	1.25	.04	.079
FS 2900	HAT-STIFFENED	235101	6.0	.07	1.5	0.80	1.25	.07	.139
		235102	6.0	.05	1.5	0.80	1.25	.06	.109
		235103	6.0	.05	1.5	0.80	1.25	.04	.089
		235104	6.0	.04	1.5	0.80	1.25	.03	.069
		235105	6.0	.04	1.5	0.80	1.25	.03	.069
		235106	6.0	.04	1.5	0.80	1.25	.03	.069
		235107	6.0	.05	1.5	0.80	1.25	.04	.089
		235108	6.0	.05	1.5	0.80	1.25	.06	.109
		235109	6.0	.07	1.5	0.80	1.25	.08	.149

PANEL DIMENSIONS:



ZEE-STIFFENED CONCEPT

HAT-STIFFENED CONCEPT

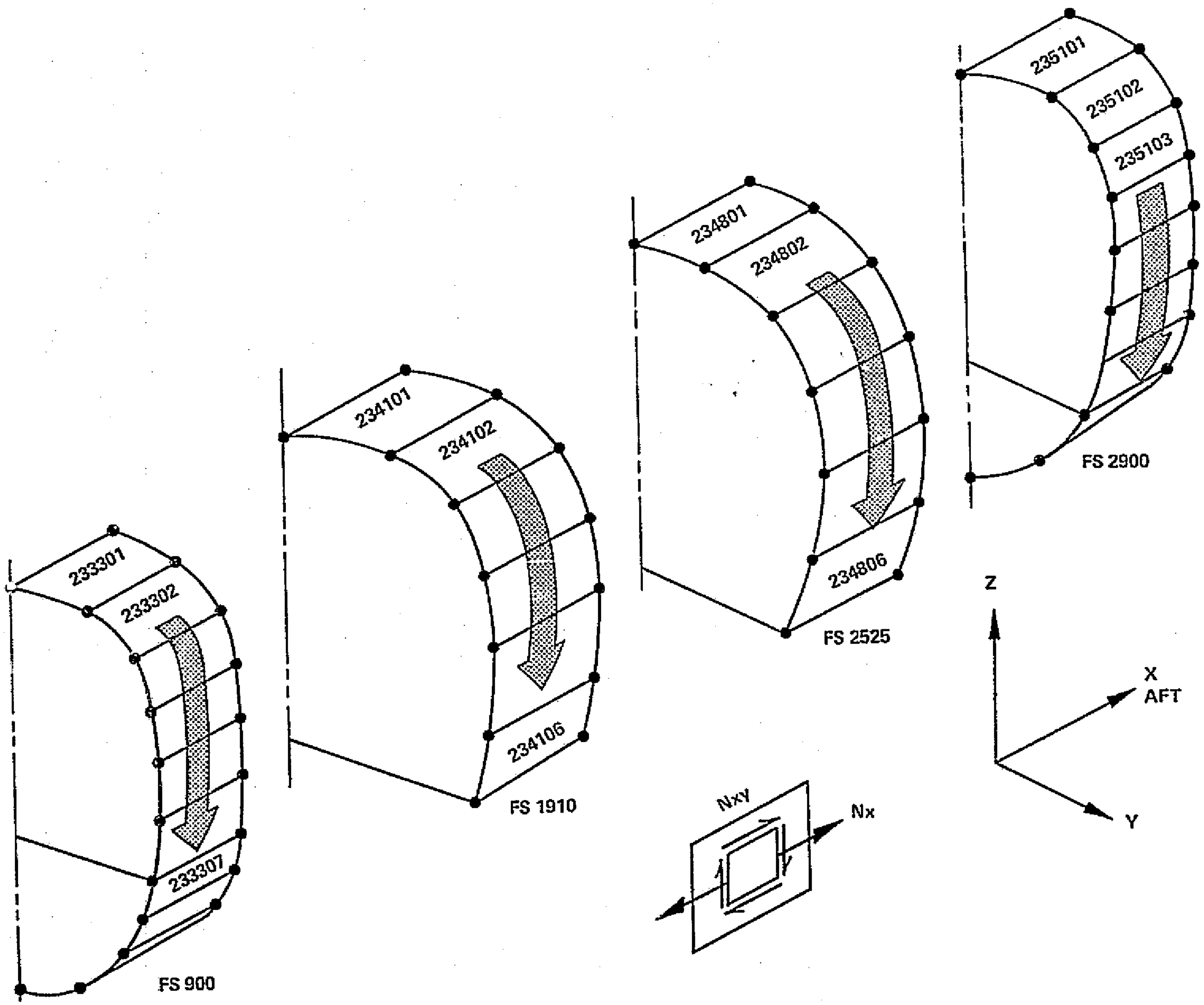


Figure 13-22. Fuselage Panel Identification - Task IIB

TABLE 13-41. REVISED FUSELAGE PANEL GEOMETRY - TASK IIB

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	FUSELAGE PANEL DIMENSIONS						
			b_s (in.)	t_s (in.)	C (in.)	f (in.)	h (in.)	t_{st} (in.)	t (in.)
FS 900	ZEE-STIFF.	–			NO CHANGES				
FS 1910	HAT-STIFF.	234103	6.0	.05	1.5	0.8	1.25	.04	.089
		234104	6.0	.05	1.5	0.8	1.25	.03	.079
FS 2525	HAT-STIFF.	234805	6.0	.08	1.5	0.8	1.25	.08	.159
		234806	6.0	.10	1.5	0.8	1.25	.08	.179
FS 2900	HAT-STIFF.	–			NO CHANGES				

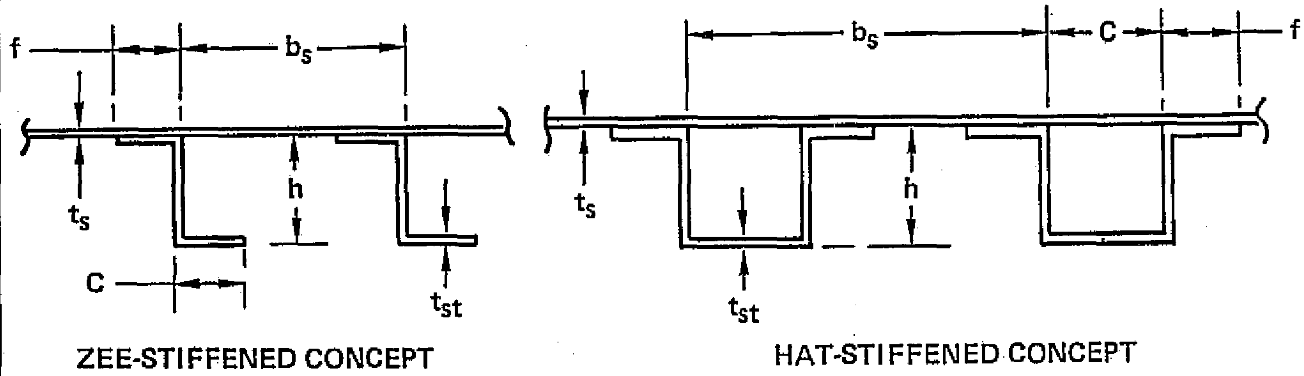


TABLE 13-42. SUMMARY OF FUSELAGE PANEL STRESSES — TASK IIB

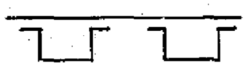
LOCATION	TYPE OF CRACK	FUSELAGE SKIN STRESS (ULT.) — ksi[1]															
		FS 900				FS 1910				FS 2525				FS 2900			
		COND	f_x	f_θ	f_{xy}	COND	f_x	f_θ	f_{xy}	COND	f_x	f_θ	f_{xy}	COND	f_x	f_θ	f_{xy}
TOP	CIRCUM.	25	34.8	—	1.77	28	57.6	—	1.14	26	52.1	—	0.53	26	39.2	—	1.61
	LONG.	22	—	34.3	—	20	—	17.1	3.23	20	—	17.1	1.17	20	—	17.1	1.19
SIDE	CIRCUM.	24	12.1	—	1.70	22	6.0	—	8.00	24	5.15	—	18.3	20	30.8	—	14.3
	LONG.	20	—	34.3	1.22	20	—	19.9	12.3	20	—	12.0	2.20	20	—	29.8	14.3
BOTTOM	CIRCUM.	—	—	—	—	—	—	—	—	—	—	—	—	20	71.4	—	4.18
	LONG.	—	—	—	—	—	—	—	—	—	—	—	—	20	—	17.1	4.18

1. LIMIT STRESS = 2/3 ULTIMATE STRESS
2. SEE SECTION 11, TABLE 11-38 FOR DEFINITION OF LOAD CONDITIONS

TABLE 13-43. FUSELAGE PANEL FAIL-SAFE ANALYSIS-
TASK IIB, POINT DESIGN REGION FS 2525

ITEM	POINT DESIGN REGION FS2525			
CRACK TYPE	LONGITUDINAL		CIRCUMFERENTIAL	
LOCATION	TOP	SIDE	TOP	SIDE
PANEL GEOMETRY				
t_s, in.	.070	.100	.070	.100
b_s, in.	–	–	6.0	6.0
b (frame spacing), in.	21.25	21.25	–	–
REINFORCEMENT AREA	(fail-safe strap)		(panel stiffener)	
Ae, in.^{2}	.06	.06	.161	.161
$\Sigma Ae/t_s$, in.	1.71	1.20	4.6	3.2
LIMIT STRESSES				
COND. NO.	20	20	26	24
f_x, ksi.	–	–	34.7	3.44
f_θ, ksi.	11.4	8.0	–	–
fxy, ksi.	0.78	1.47	0.35	12.2
FRACTURE TOUGHNESS				
ko, ksi - $\sqrt{in.}$	–	–	130	138
1/2 ko, ksi - $\sqrt{in.}$	65	69	–	–
CRACK GEOMETRY				
LENGTH (L), in.	42.5	42.5	12.0	12.0
$\sqrt{L}$, $\sqrt{in.}$	6.52	6.52	3.46	3.46
REINFORCEMENT EFF.				
γ	1.75	1.70	1.61	1.53
ψ	0.99	0.97	1.00	0.40
ALLOWABLE STRESSES				
$Fg = \gamma\psi\, ko/\sqrt{L}$	17.3	17.4	60.4	24.4
F_s = 1/2 Fg	8.64	8.73	30.2	12.3
MARGINS OF SAFETY				
Axial (MS = Fg/f - 1)	+0.52	+1.18	+0.74	+HIGH
Shear	+HIGH	+HIGH	+HIGH	+0.01

TABLE 13-44. FUSELAGE PANEL FAIL-SAFE WEIGHT PENALTY — TASK IIB, POINT DESIGN REGION FS 2525

REGION/CONCEPT	PANEL LOCATION	FAIL-SAFE REQUIREMENTS					PANEL CIRCUM C_i (in.)
		PANEL $\bar{t}$ (in.)	STRAP			$\Sigma\bar{t}$ (in.)	
			A (in^2)	b (in.)	$\bar{t}$ (in.)		
FS 2525	234801 (TOP)	—	0.060	21.25	0.0028	0.0028	39.64
HAT-STIFFENED PANEL CONCEPT	234802	—	0.060	21.25	0.0028	0.0028	29.72
	234803	—	0.060	21.25	0.0028	0.0028	23.68
	234804	—	0.140	21.25	0.0066	0.0066	17.86
	234805	0.090	0.060	21.25	0.0028	0.0928	11.82
	234806 (SIDE)	0.100	0.060	21.25	0.0028	0.1028	11.90
	AVERAGE VALUES	$\bar{t}$ = 0.020 in.; W = 0.46 lb./sq. ft.					

NOMENCLATURE:

$\bar{t}$ = EQUIVALENT SURFACE PANEL THICKNESS

A = AREA OF FAIL-SAFE STRAP

b = FRAME SPACING

A/b = $\bar{t}$ (STRAP)

$\Sigma\bar{t} = \bar{t}\,(\text{PANEL}) + \bar{t}\,(\text{STRAP})$

C_i = PANEL CIRCUMFERENCE

$$\bar{t}\,(\text{AVG}) = \sum_{i=1}^{6} C_i\bar{t}_i \Big/ \sum_{i=1}^{6} C_i$$

$$W\,(\text{AVG}) = 23.04 \times \bar{t}\,(\text{AVG})$$

are calculated for the entire point design region, the respective values for FS 2525 are 0.020 inches and 0.46 lb/sq ft. The added panel thickness requirements reflect the revised geometry shown in Table 13-41. Table 13-45 summarizes the results of the Task II fuselage fail-safe analyses. This table presents a summary of the pertinent data derived from the detail calculations, indicates the margin of safety and weight penalty associated with the specific panels, and the average weight penalty for the entire point design region. All regions required additional structure to meet the fail-safe requirements. The highest weight penalty, 0.46 lb/sq. ft., was associated with the mid-body region at FS 2525. The aftbody region at FS 2900 exhibited the highest fail-safe capability i.e., lowest weight penalty, 0.10 lb/sq. ft.

In general, selective panel stiffening was required to meet the circumferential crack criteria; whereas, all regions required circumferential fail-safe straps to attain the longitudinal crack criteria.

TABLE 13-45. SUMMARY OF FUSELAGE FAIL-SAFE ANALYSES — TASK IIB

DESIGN CONCEPT	POINT DESIGN REGION	PANEL LOCATION	TYPE OF CRACK	CRACK LENGTH (in.)	$\Sigma A_e/t$ (1)(2) (in.)	REINFORCEMENT EFFICIENCIES		MARGIN OF SAFETY	WEIGHT PANELTY (ΔW) (lb./sq. ft.)	
						γ	ψ		PANEL	POINT DESIGN REGION
ZEE-STIFF.	FS 900	TOP	CIRCUM	8.0	2.0	1.36	1.00	+1.25	NONE	(0.25)
		SIDE	CIRCUM	8.0	2.0	1.36	0.98	+ HIGH	NONE	
		TOP	LONG	21.5	7.2	1.97	1.00	+0.01	0.25	
		SIDE	LONG	21.5	7.2	1.97	0.99	+0.01	0.25	
HAT-STIFF. CONCEPT	FS 1910	TOP	CIRCUM	12.0	3.5	1.55	1.00	+0.51	NONE	(0.22)
		SIDE	CIRCUM	12.0	4.1	1.57	0.58	+2.11	NONE	
		TOP	LONG	44.3	1.7	1.75	0.97	+0.46	0.06	
		SIDE	LONG	44.3	5.0	2.20	0.80	+0.02	0.15	
HAT-STIFF. CONCEPT	FS 2525	TOP	CIRCUM	12.0	4.6	1.61	1.00	+0.74	NONE	(0.46)
		SIDE	CIRCUM	12.0	3.2	1.53	0.40	+0.01	2.34	
		TOP	LONG	42.5	1.7	1.75	0.99	+0.52	0.06	
		SIDE	LONG	42.5	1.2	1.70	0.97	+1.18	0.06	
HAT-STIFF. CONCEPT	FS 2900	TOP	CIRCUM	12.0	4.1	1.58	1.00	+1.27	NONE	(0.10)
		SIDE	CIRCUM	12.0	3.2	1.53	0.87	+1.11	NONE	
		BOTTOM	CIRCUM	12.0	4.6	1.60	0.99	+0.25	NONE	
		TOP	LONG	42.0	1.7	1.70	1.00	+0.50	0.07	
		SIDE	LONG	42.0	10.0	2.77	0.86	+0.04	0.22	
		BOTTOM	LONG	42.0	1.7	1.70	0.96	+0.44	0.07	

NOTES:

1. ΣA_e = SUM OF EFFECTIVE AREAS
2. t = SKIN THICKNESS
3. PANEL WEIGHT PENALTY

$$\Delta\bar{t}_i = \Delta\bar{t}_{PANEL} + A_{STRAP}/b$$
$$\Delta W_i = 144\,\rho\Delta\bar{t}_i = 23.04\,\Delta\bar{t}_i$$

POINT DESIGN REGION WEIGHT PENALTY

$$\Delta\bar{t}_{AVG.} = \Sigma C_i \Delta\bar{t}_i / \Sigma C_i$$
$$\Delta W = 23.04\,\Delta\bar{t}_{AVG.}$$

WHERE:

$\Delta\bar{t}_i$ = EQUIVALENT SURFACE PANEL THICKNESS OF i^{th} PANEL

$\Delta\bar{t}_{AVG}$ = AVERAGE SURFACE PANEL THICKNESS OF FUSELAGE CROSS-SECTION

C_i = CIRCUMFERENCE OF i^{th} SURFACE PANEL

$\Delta\bar{t}_{PANEL}$ = ADDITIONAL THICKNESS OF i^{th} PANEL FOR FAIL-SAFE

A_{STRAP} = STRAP AREA OF i^{th} PANEL FOR FAIL-SAFE

b = FRAME SPACING

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SECTION 14

ACOUSTICS

By

G. W. Davis, W. A. Guinn, W. L. LaBarge

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List of Symbols

A	Cross-sectional Area
a,b	Panel dimension between supported edges
BL	Butt line
b_s	Stiffener pitch
dB	Measure of the intensity of pressure fluctuations, decibels
FS	Fuselage station
f	Frequency
Hz	Hertz
I	Moment of inertia per unit width, $in.^3$
L	Length in direction of maximum stiffness
OASPL	Overall sound pressure level
Sn	Strouhal number
t	Thickness
$\bar{t}$	Equivalent panel thickness
V	Velocity
w	Equivalent panel weight, lb/sq. ft.
Z	Distance from neutral axis to outermost fiber
ΔdB	Incremental change in pressure intensity

SECTION 14

ACOUSTICS

INTRODUCTION

Experience gained in the development of aircraft structural design has demonstrated the importance of a coordinated design program in which sonic fatigue prevention plays an integral part.

The principal components of any sonic fatigue prevention program are: (1) a definition of the aircraft's acoustic environment; and (2) the design of structure which will withstand the acoustically induced loads without fatigue cracking. Each of these aspects involves a combined analytical and experimental approach.

Due to the preliminary nature of this investigation, only a preliminary assessment of the sonic fatigue capability of the structural configurations could be ascertained for the supersonic cruise aircraft. This assessment was conducted in the following steps:

- The acoustic environment was estimated for the baseline airplane during take-off.
- The methods of analysis and associated design charts were defined for use in the detail analysis.
- Selective wing and fuselage surface panels were analyzed to assess the relative merit of the structural concepts.

The results of this assessment are reported in the following text under the titles: sonic environment, methods, and analysis. For continuity, the results of the Task I analysis (wing and fuselage) are presented in their entirety followed by the results of the Task II analysis.

SONIC ENVIRONMENT

The acoustic environment to which the baseline airplane is subjected during takeoff was estimated from empirical free field acoustic levels generated by an existing turbojet engine. The engine selected for the supersonic cruise airplane study is a Mach 2.7 duct burning turbofan engine, designated the BSTF 2.7-2.

A schematic drawing of this engine is shown in Figure 14-1 with the engine parameters listed in Table 14-1. The acoustic environment generated by the baseline turbofan engine was estimated by adjusting the empirical acoustic levels to account for the differences in the geometric characteristics of the engines, the operating parameters and the presence of structure within the acoustic field.

In support of the Propulsion-Airframe Integration Study, reported in Section 19, the acoustic environment in terms of overall sound pressure levels (OASPL) and octave band levels was defined for the same engine mounted further forward than the baseline location. The engine location for these two designs are presented in Figure 14-2.

Reference Contours

The acoustic environment is based on jet near-field noise prediction methods given by Franken and Kerwin in Reference 1. The basis for the noise contours of this study was the acoustic levels defined by the above investigators for an existing turbojet engine, Figure 14-3. These data reflect a circular nozzle with an exhaust area of 0.66 sq. ft. and an exhaust velocity of 1850 ft. per sec. These referenced contours were extrapolated and scaled to the baseline airplane dimensions, Figure 19-1. The scaled contours were then adjusted to reflect the differences in:

- Engine geometric characteristics.
- Operating parameters.
- Presence of structure within the acoustic field.

CONCEPTUAL GAS PATH SCHEMATIC
FAN PRESSURE RATIO = 3.0

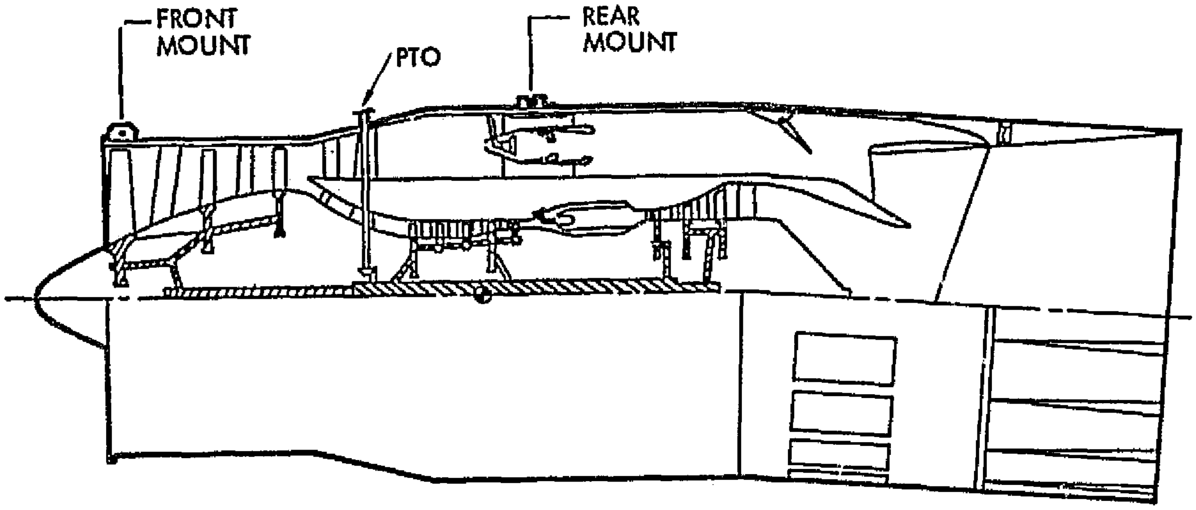


Figure 14-1. Duct Burning Turbofan Engine - Mach 2.7

TABLE 14-1. PROPULSION SYSTEM PARAMETERS

Engine: Number of engines: Noise suppression: Inlet/nozzle: Thrust/weight – (lift off): Lift off Speed:	BSTF 2.7-2 duct burning turbofan 4 FAR 36-5 Axisymmetric/variable convergent-divergent 0.36 Mach 0.30	
Scale Factor:	1.0 (Ref.)	1.147
Net thrust, lb. (A)	78,000	89,466
Engine weight, lb. (B)	11,143	12,781
ACAP, ft^2	33.1	38.0
DMAX, in.	90	96.4
DCOMP, in.	79.4	85.0
DNOZ, in.	90	96.4
LENG, in.	255	267.5
LINLET, in.	189.3	203.9
Study Application	Task I	Task II

(A) SLS, Max. Power, uninstalled
(B) Includes reverser and suppressor

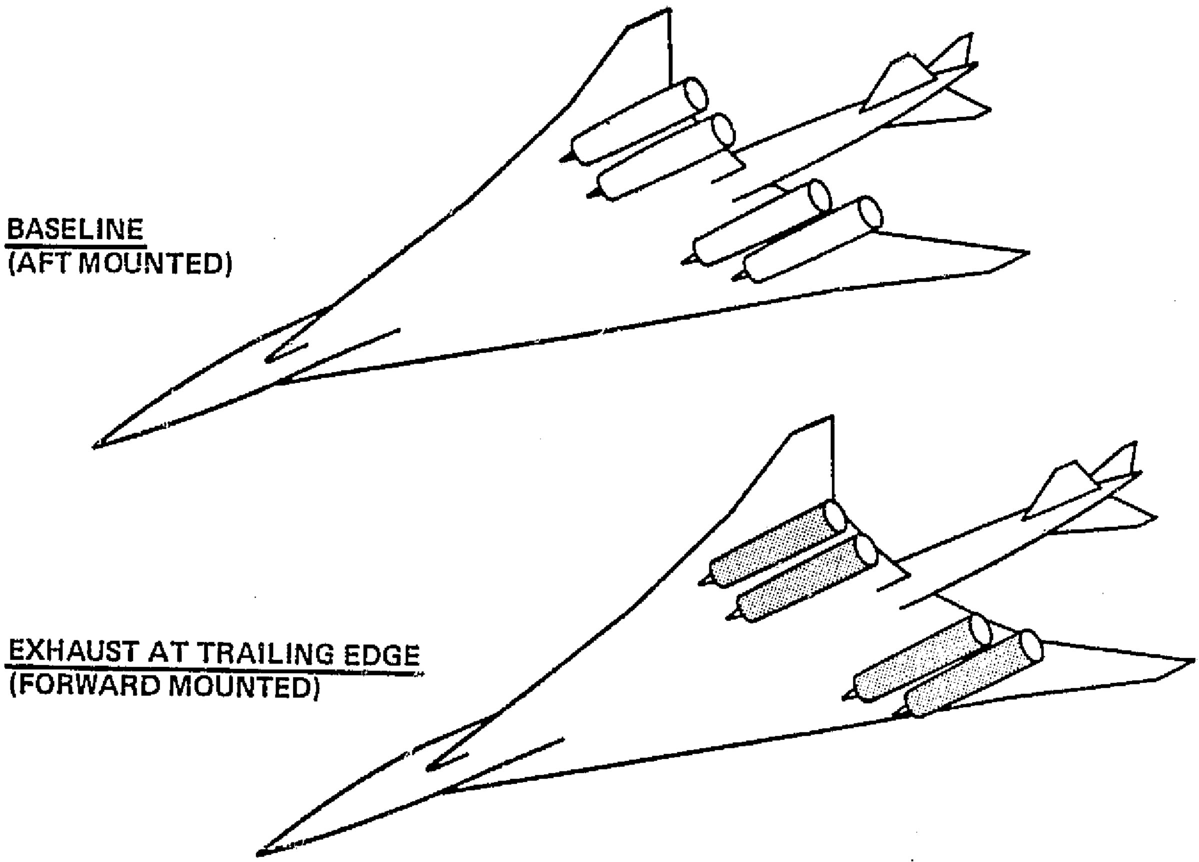


Figure 14-2. Candidate Engine Location

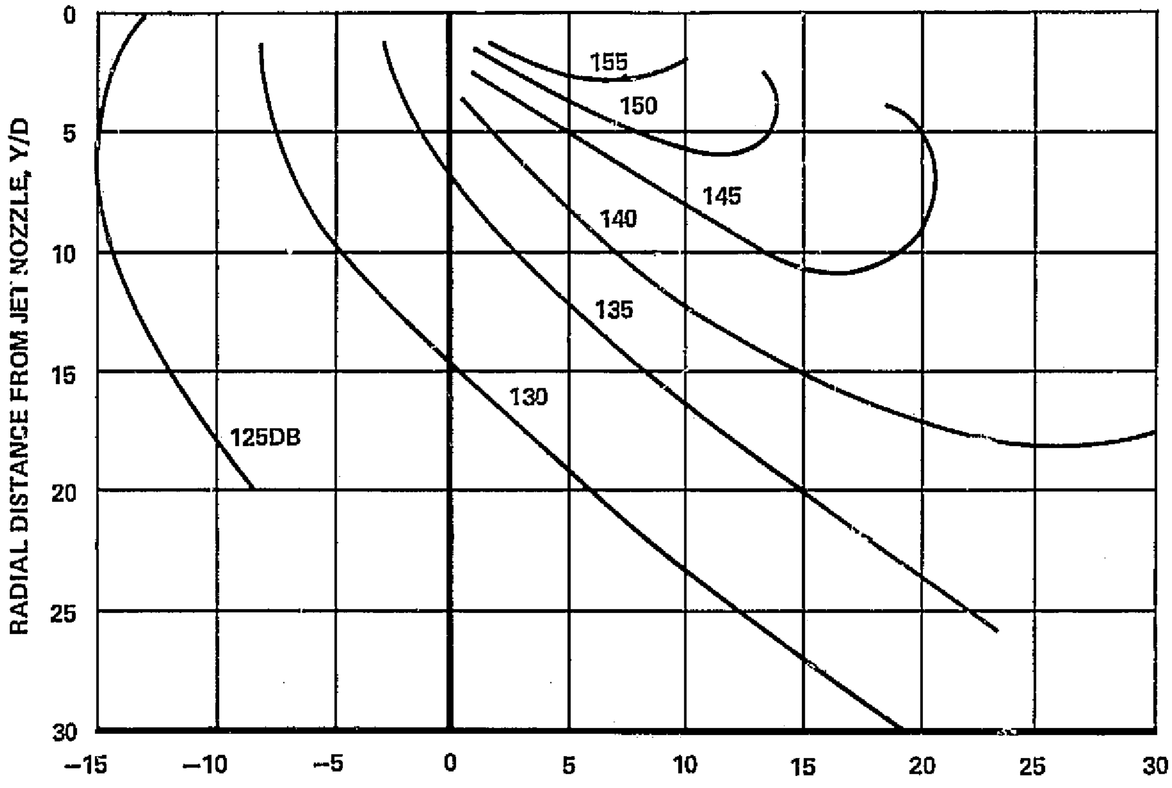


Figure 14-3. Near-Field Noise Contours - Reference Turbo-Jet Engine

Engine Characteristics

The following table defines the engine nozzle area and exhaust velocity of the referenced engine and the turbofan engine used for the study.

ENGINE	NOZZLE PARAMETERS	
	Exhaust Area ($Ft.^2$)	Exhaust Velocity (Ft/Sec.)
Referenced turbojet	0.66	1850
Duct burning turbofan	21.4	2370

Using these data, the values for the referenced contours were scaled to account for the difference in nozzle area and velocity. The nozzle area relationship is directly proportional to the area ratio; whereas, the velocity relationship is proportional to the velocity ratio to the eighth power. The change in the noise contours attributed to the differences in exhaust velocity was calculated as follows:

$$\Delta dB = 80 \log_{10} \frac{2370}{1850} = 8.64 \text{ dB}$$

In addition, the noise suppressor attentuation was estimated from the predicted values given by General Electric for their AST engine, Reference 2. For the BSTF 2.7-2 duct burning turbofan with 2400 ft. per sec. exhaust velocities the estimated noise attenuation is 14.5 dB. In summary, the incremental changes attributed to the engine geometry and noise suppressor are shown in the following table.

ITEM	Changes Over Reference Engine Noise Contours
Nozzle area	+ 15.12 dB
Velocity ratio	+ 8.64 dB
Noise suppressor	- 14.50 dB
ΔdB	+ 9.26 dB

From these results, 10 dB were added to each contour shown in Figure 14-3.

Addition of Noise

The noise contours for each engine overlapped the noise of the other engines. Therefore, the noise at a given point was determined by adding the noise in pairs. To expedite these logarithmic calculations Table 14-2 was used. An explanation of the use and the limiting conditions are included on this table.

Structure Within the Field

To account for the presence of structure within the acoustic field, the reflected acoustic wave was assumed to cause a pressure doubling near aircraft surfaces. Therefore, 6 dB was added to the noise at each point.

Isointensity Contours

The OASPL was determined using the reference contours (Figure 14-3) and the calculated incremental changes associated with the AST design. Isointensity contours were defined for each of the engine locations (Figure 14-2) after OASPL had been defined at sufficient points. Figure 14-4 displays the isointensity contours for the baseline engine location and Figure 14-5 presents the corresponding contours for the forward mounted engine.

The peak frequency was determined by consideration of the Strouhal number, ($S_n = fD/V$). Assuming a Strouhal number of 0.3, commensurate with a supersonic jet with a 54 tube nozzle, and solving the equation explicitly for the frequency f, a peak frequency of 995 Hz is obtained. The calculations are as follows:

TABLE 14-2. COMBINING SOUND PRESSURE LEVELS IN DB'S

LIMITATIONS: USE OF THIS TABLE LIMITED TO THE TWO FOLLOWING CONDITIONS:

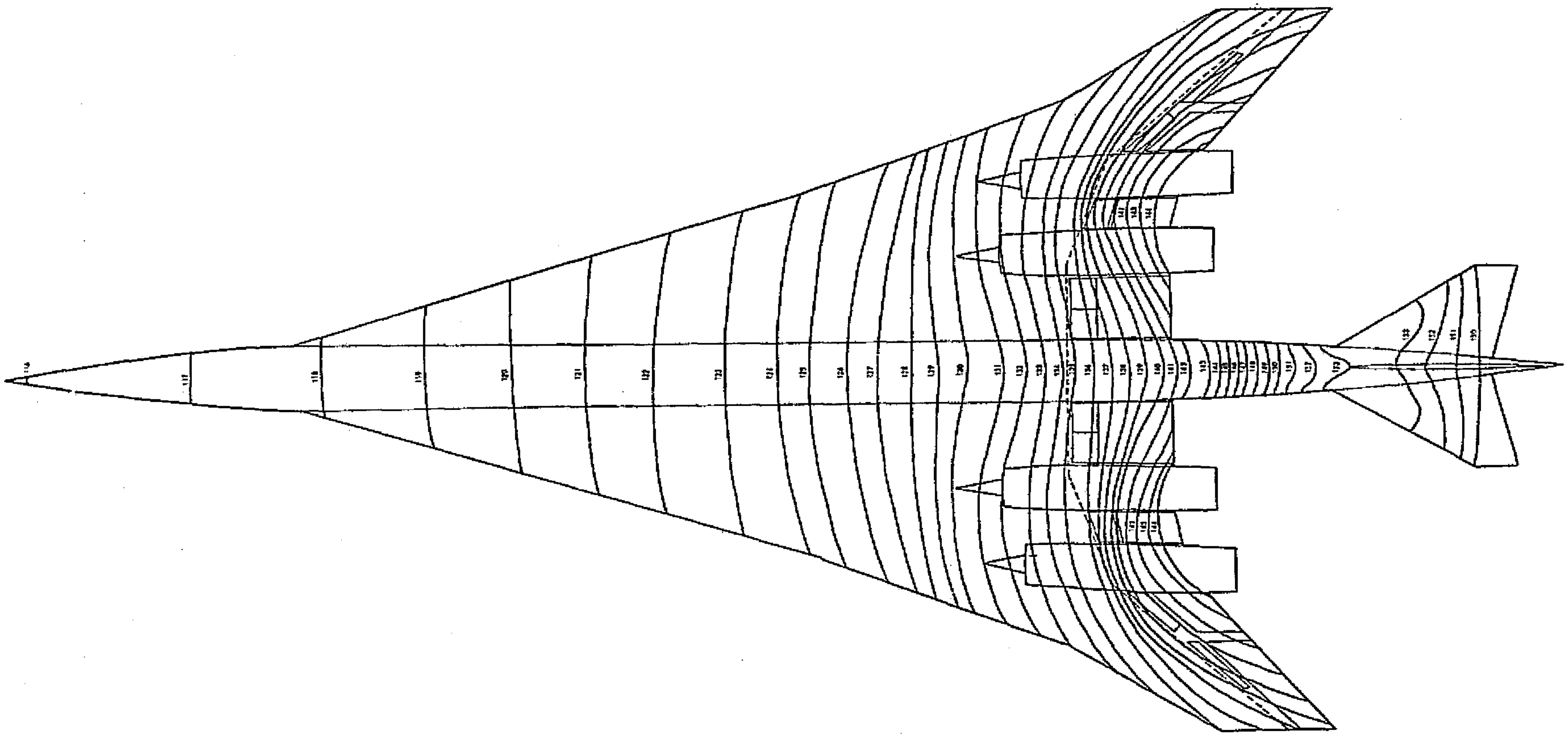
A. COMBINATIONS OF SINE WAVES NO TWO OF WHICH HAVE THE SAME FREQUENCY.

B. ANY COMBINATION OF RANDOM NOISE SOURCES, WITH OR WITHOUT COMBINATIONS OF SINE WAVES.

EXPLANATION OF TABLE: THE GROUP OF NUMBERS BENEATH THE BOLD NUMERAL AT THE TOP OF EACH BOX, REPRESENTS THE DIFFERENCE IN DB, BETWEEN ANY TWO SOUND PRESSURE LEVELS, L_1 & L_2. ($L_1 \geq L_2$)

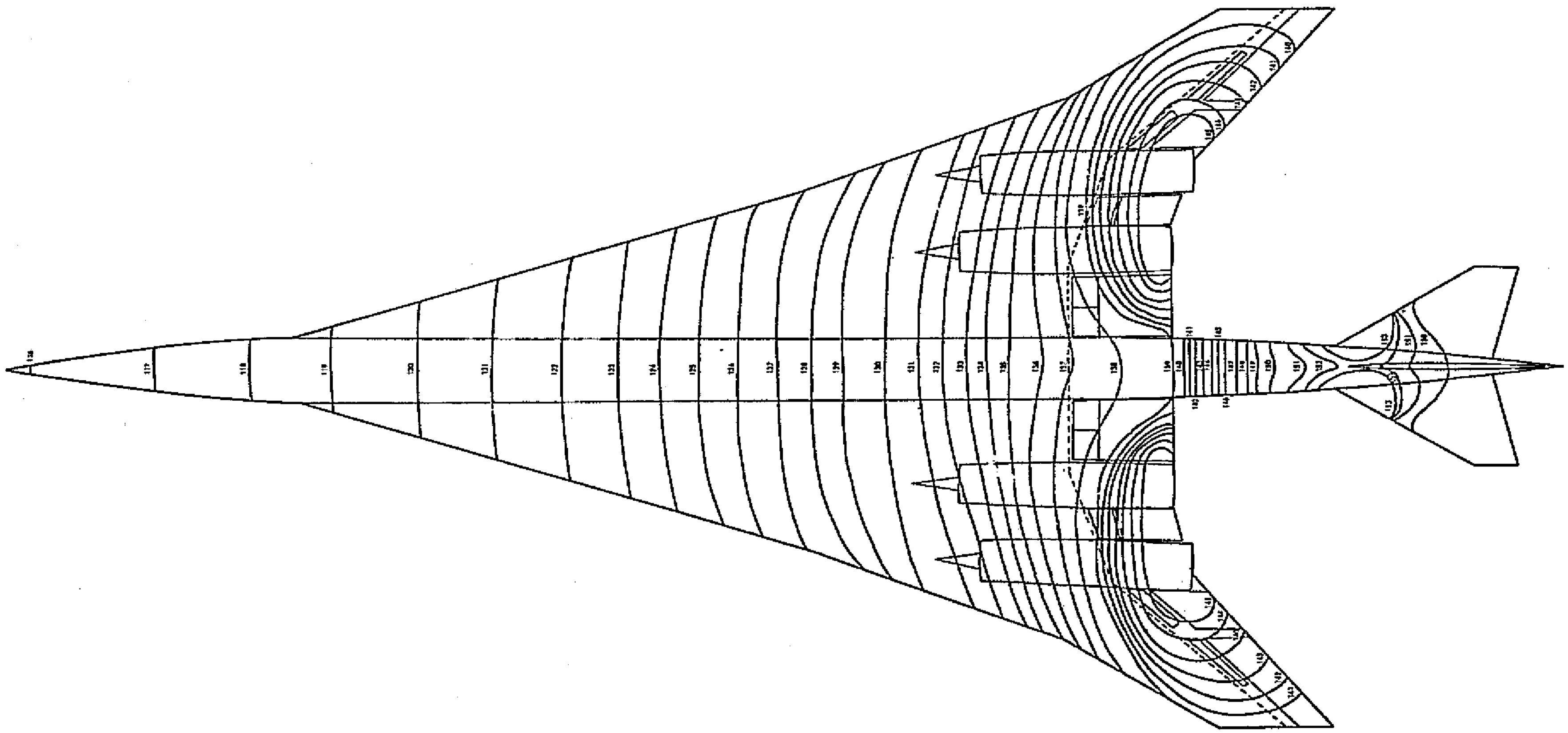
THE VALUES IN THE RIGHT HAND COLUMN OF EACH BOX ARE THE NUMBER OF DB TO BE ADDED TO L_1 TO OBTAIN THE RESULTANT OF L_1 & L_2 IN DB.

Box	Difference (dB)	Add to L_1 (dB)
0	0.0, 0.1	3.0
	0.2, 0.3	2.9
	0.4, 0.5	
	0.6, 0.7	2.7
	0.8, 0.9	2.6
1	1.0, 1.1	2.5
	1.2, 1.3, 1.4	2.4
	1.5, 1.6, 1.7	2.3
	1.8, 1.9	2.2
2	2.0, 2.1, 2.2	2.1
	2.3, 2.4, 2.5	2.0
	2.6, 2.7	1.9
	2.8, 2.9	1.8
3	3.0, 3.1	1.8
	3.2, 3.3	1.7
	3.4, 3.5, 3.6, 3.7	1.6
	3.8	1.5
4	4.0, 4.1	1.5
	4.2, 4.3	1.4
	4.4, 4.5, 4.6, 4.7	1.3
	4.8, 4.9	1.2
5	5.0, 5.1	1.2
	5.2, 5.3, 5.4, 5.5, 5.6	1.1
	5.7, 5.8, 5.9	1.0
6	6.0, 6.1	1.0
	6.2, 6.3, 6.4, 6.5, 6.6	0.9
	6.7, 6.8, 6.9	0.8
7	7.0, 7.1, 7.2	0.8
	7.3, 7.4, 7.5, 7.6, 7.7, 7.8, 7.9	0.7
8	8.0, 8.1, 8.2, 8.3, 8.4, 8.5, 8.6	0.6
	8.7, 8.8, 8.9	0.5
9	9.0, 9.1, 9.2, 9.3, 9.4, 9.5, 9.6	0.5
	9.7, 9.8, 9.9	0.4
10	10.0, 10.1, 10.2, 10.3, 10.4, 10.5, 10.6, 10.7	0.4
	10.8, 10.9	0.3
11	11.0, 11.1, 11.2, 11.3, 11.4, 11.5, 11.6, 11.7, 11.8, 11.9	0.3
12	12.0, 12.1, 12.2, 12.3, 12.4	0.3
	12.5, 12.6, 12.7, 12.8, 12.9	0.2
13	13.0, 13.1, 13.2, 13.3, 13.4, 13.5, 13.6, 13.7, 13.8, 13.9	0.2
14	14.0, 14.1, 14.2, 14.3, 14.4, 14.5	0.2
	14.6, 14.7, 14.8, 14.9	0.1
15	15.0, 15.1, 15.2, 15.3, 15.4, 15.5, 15.6, 15.7, 15.8, 15.9	0.1



OVERALL SPL REF. .0002 μBAR

Figure 14-4. Overall Sound Pressure Level - Baseline Engine Location



OVERALL SPL REF. .0002 μBAR

Figure 14-5. Overall Sound Pressure Level - Forward Mounted Engine

$$\text{Tube area} = \frac{21.4}{54} = 0.396 \text{ ft}^2$$

$$\text{Tube Diameter} = \sqrt{\frac{4x.396}{3.14}} = 0.712 \text{ ft.}$$

and from the Strouhal number equation

$$f = \frac{.30 \times 2370}{.712} = 995 \text{ Hz}$$

The spectrum shape was then determined by comparison of spectra from several suppressor nozzles and is shown in Figure 14-6. The octave band noise contours at the center frequency were determined by subtracting the values shown in Figure 14-6 from the OASPL displayed in Figures 14-4 and 14-5. In tabular form these values are:

FREQUENCY (Hz)	Δ dB
63	-15
125	-14
250	-12
500	-10
1000	- 6

Figures 14-7 through 14-11 present the noise contours for the baseline engine placement for octave band levels with 63, 125, 250, 500, and 1000 Hz center frequencies respectively. Figure 14-12 through 14-16 give the corresponding noise contours for the forward mounted engine.

METHOD OF ANALYSIS

The two most important properties of structure from the standpoint of sonic fatigue resistance are: (1) its resonant frequencies; and (2) its "quality of detail design".

The first of these is important because excitation of a structure at its resonant frequency can induce stresses in the structure on the order of 50 times as great as those which would result from the same load applied statically.

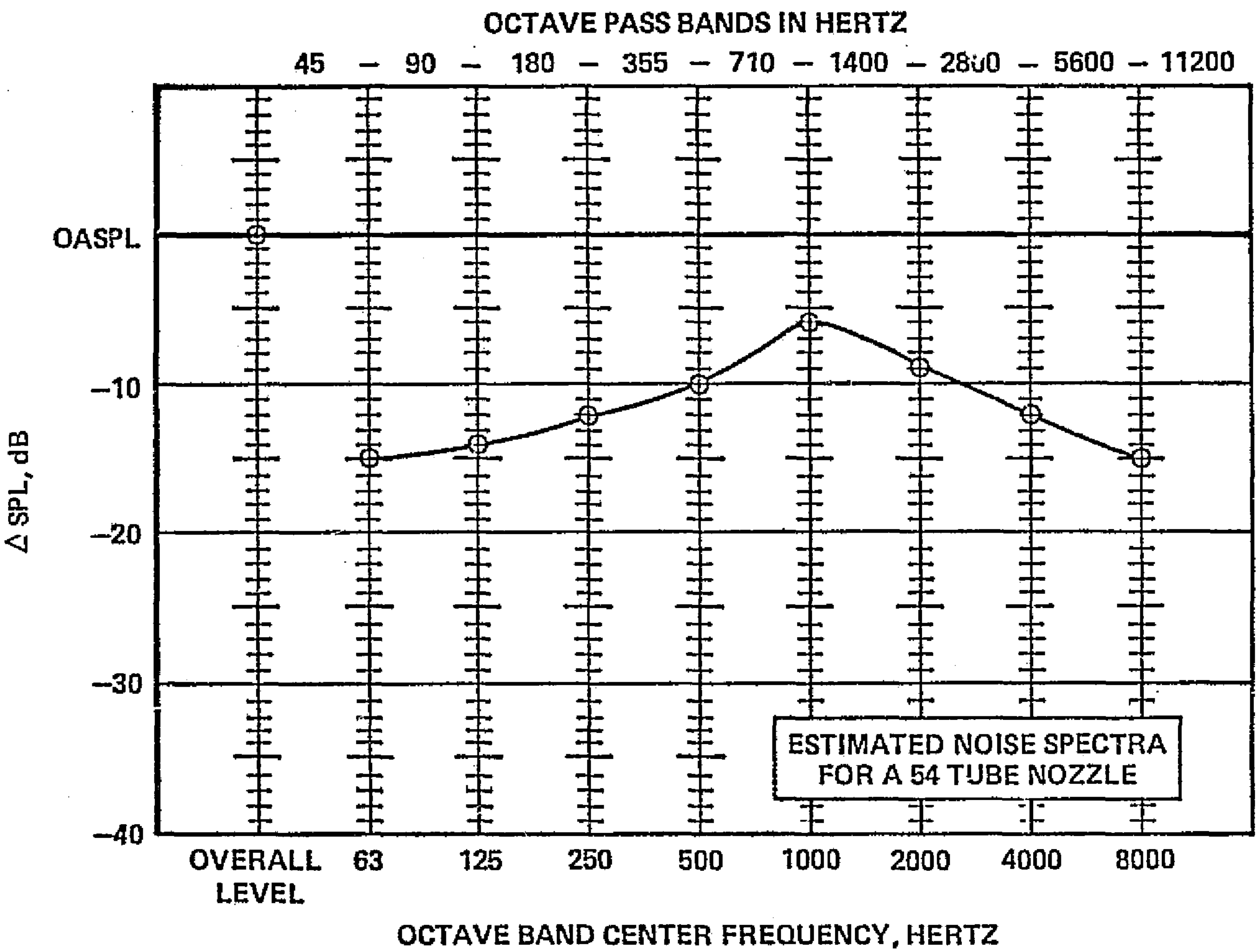


Figure 14-6. Exhaust Noise Spectra

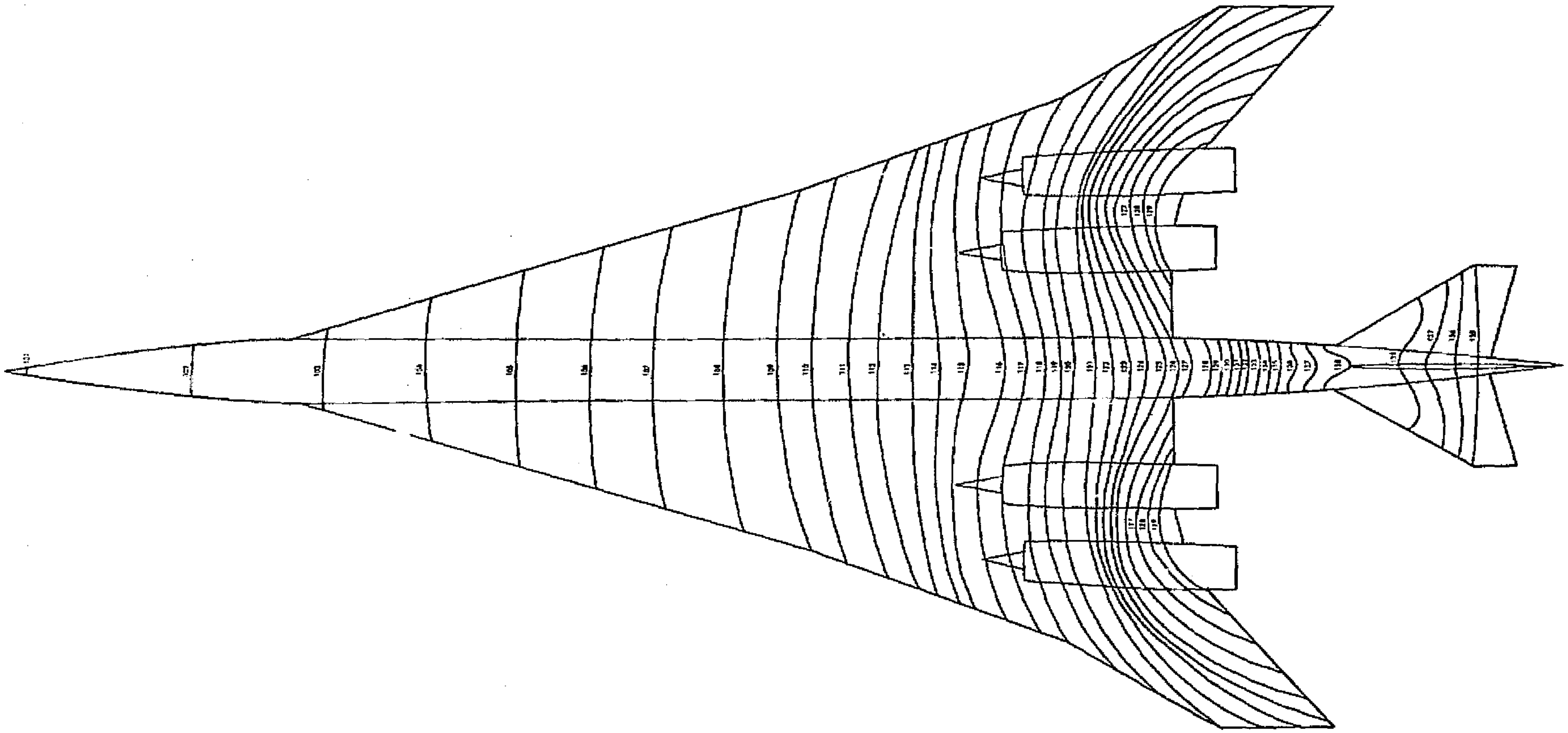


Figure 14-7. 63 Hz. Octave Band Pressure Levels - Baseline Engine Location

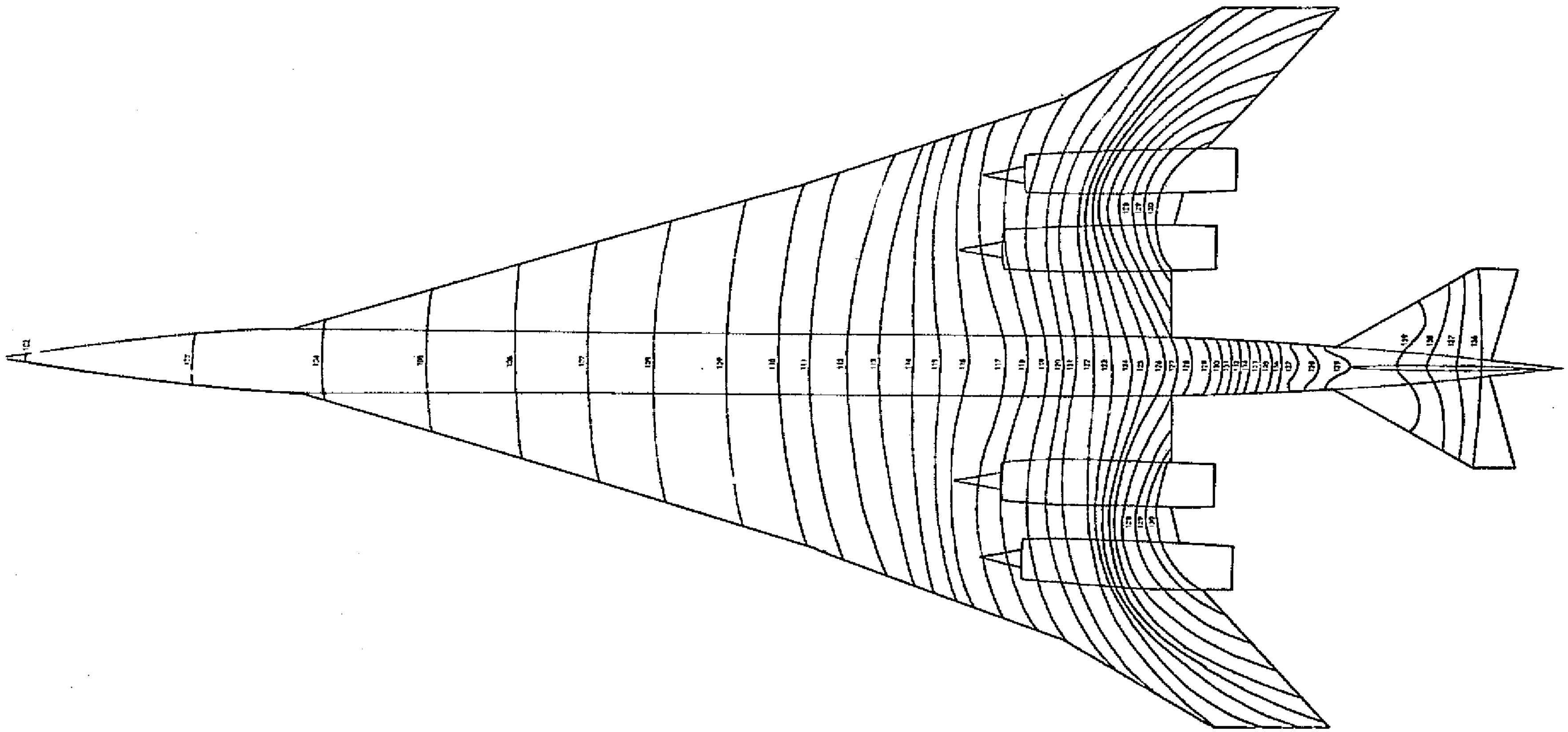


Figure 14-8. 125 Hz. Octave Band Pressure Levels - Baseline Engine Location

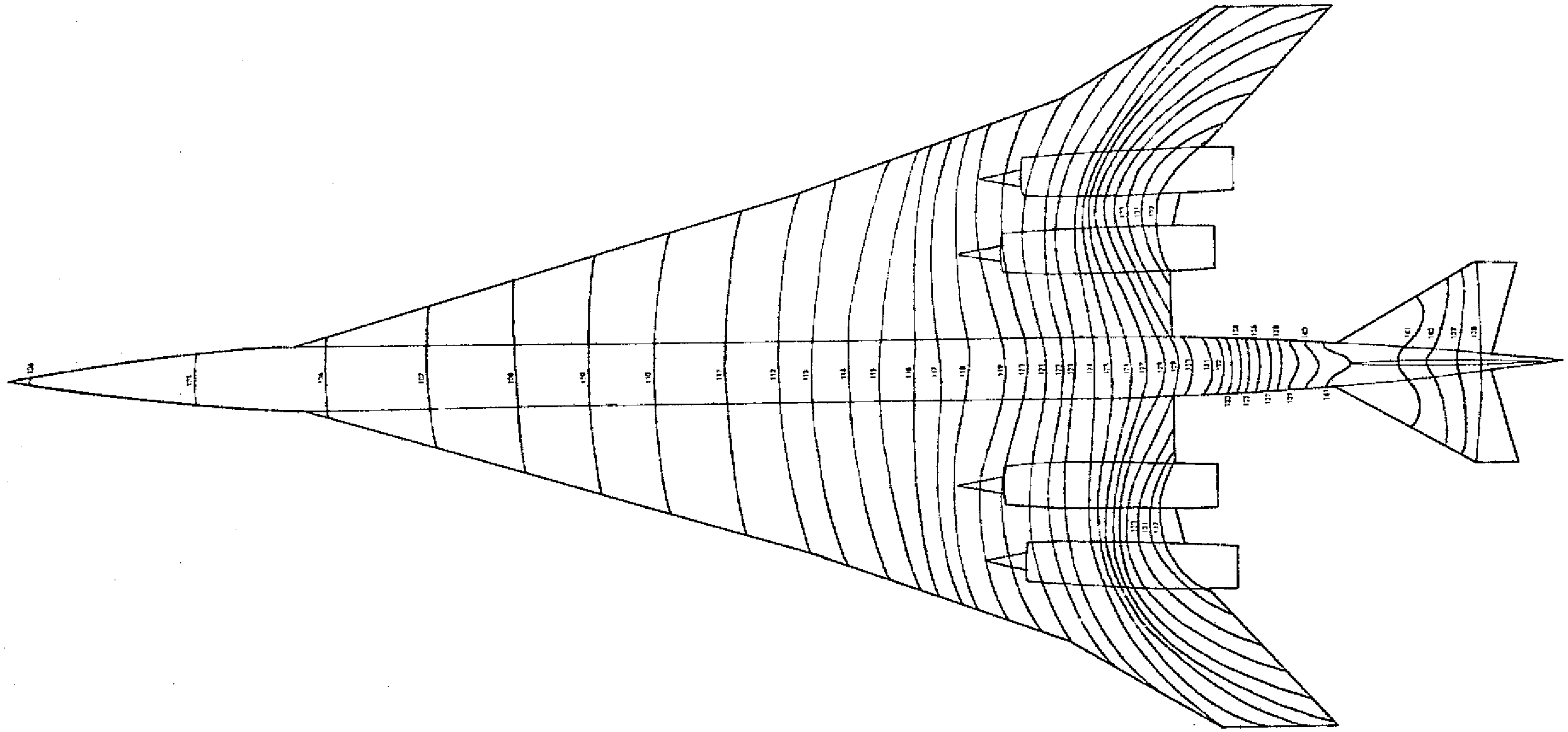


Figure 14-9. 250 Hz. Octave Band Pressure Levels - Baseline Engine Location

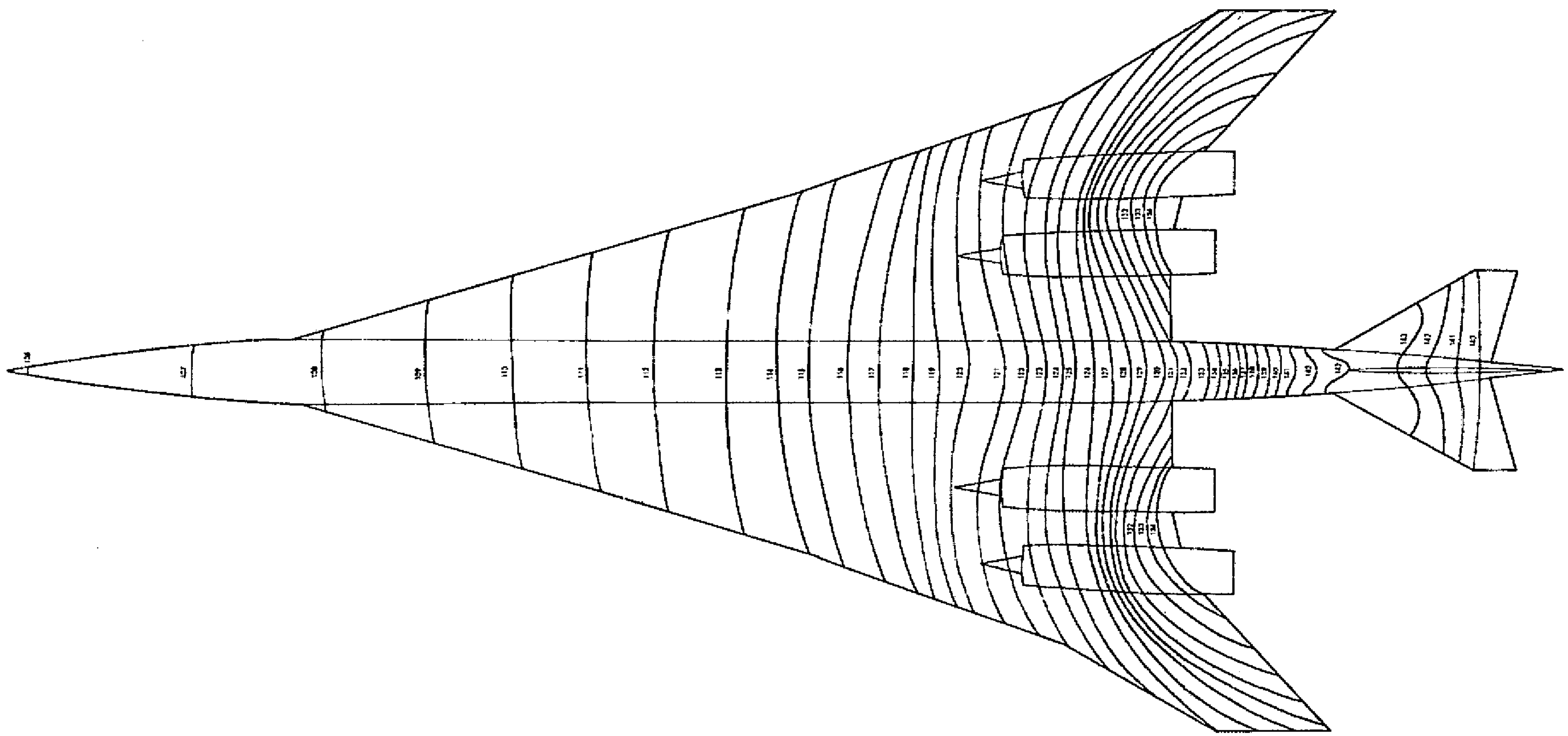


Figure 14-10. 500 Hz. Octave Band Pressure Levels - Baseline Engine Location

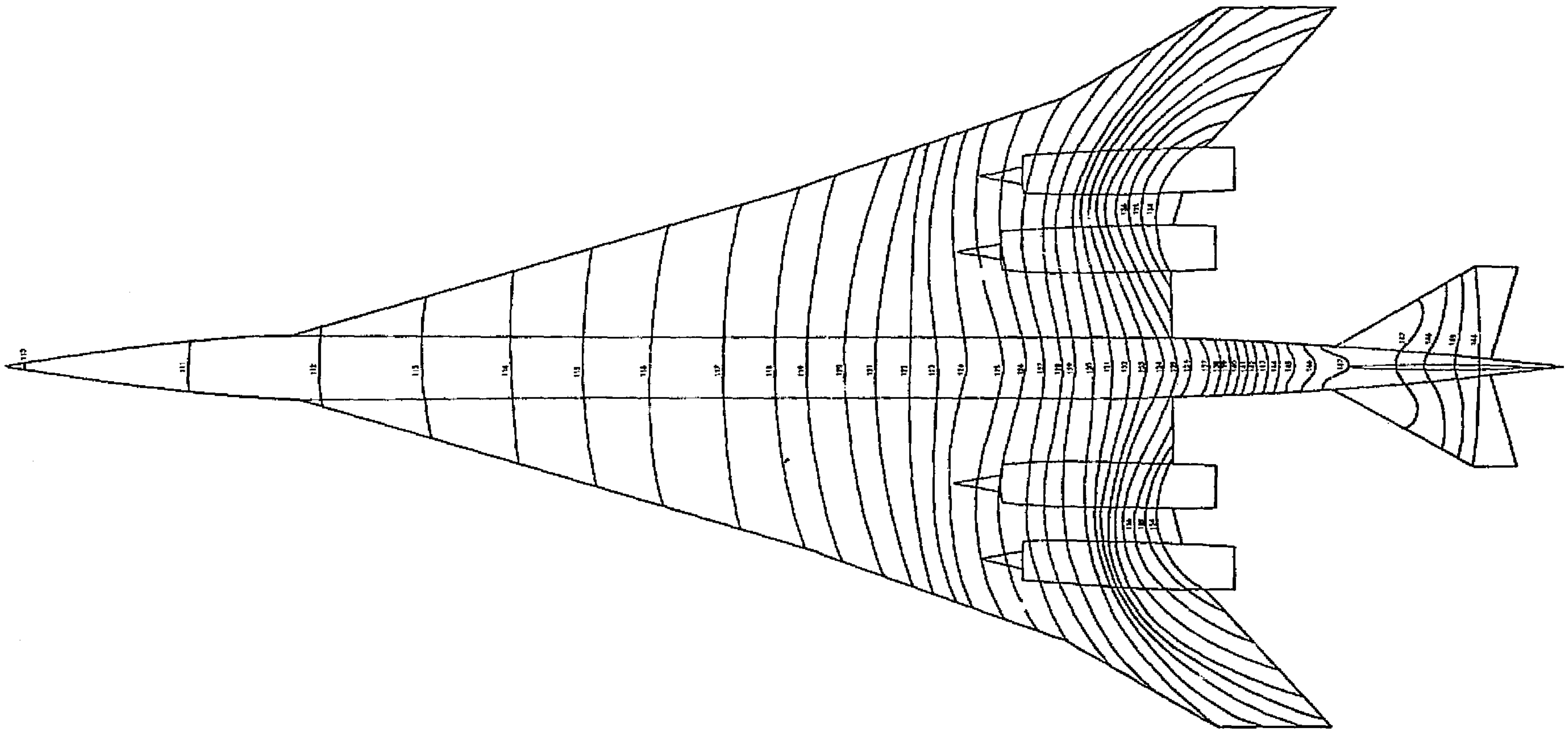


Figure 14-11. 1000 Hz. Octave Band Pressure Levels - Baseline Engine Location

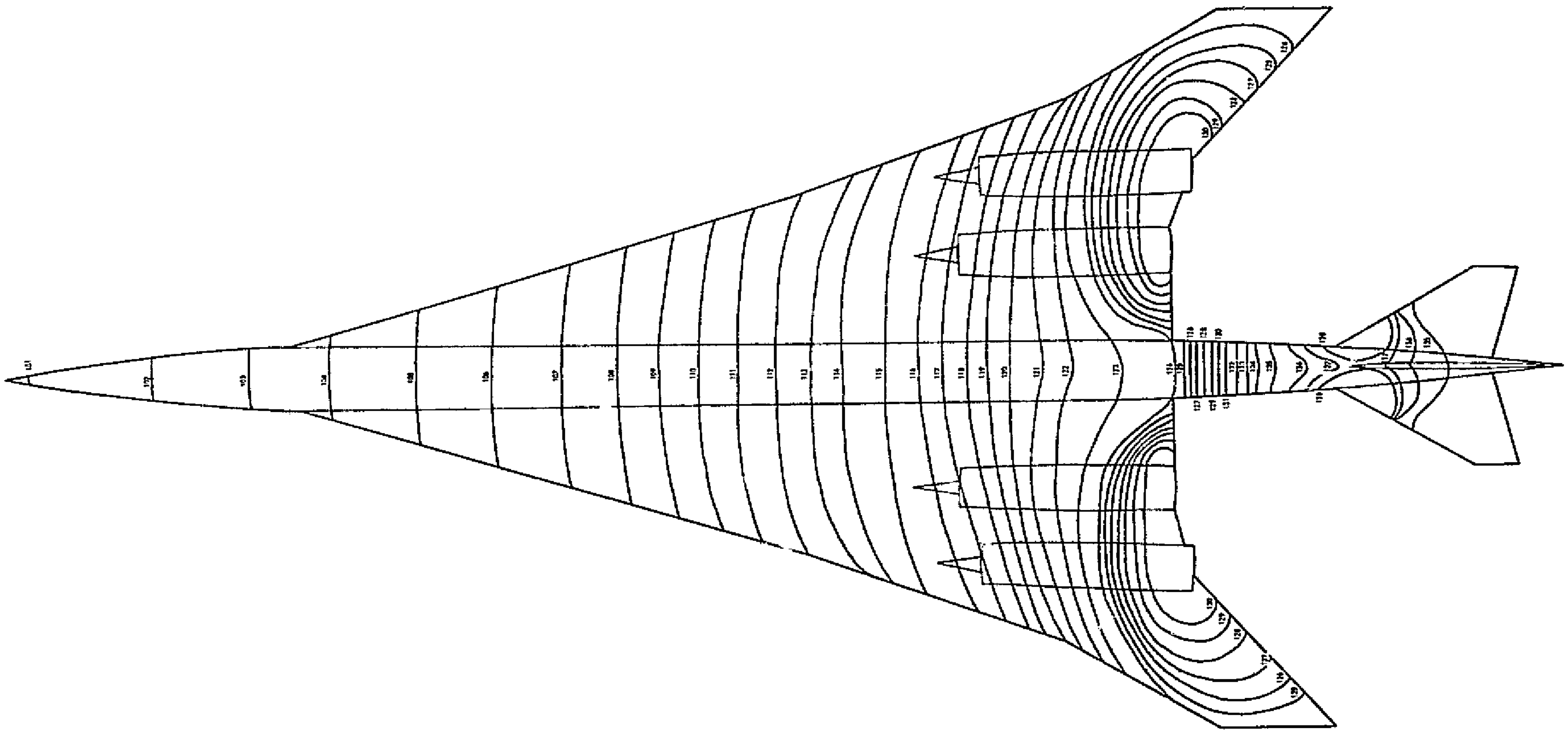


Figure 14-12. 63 Hz. Octave Band Pressure Levels - Forward Mounted Engine

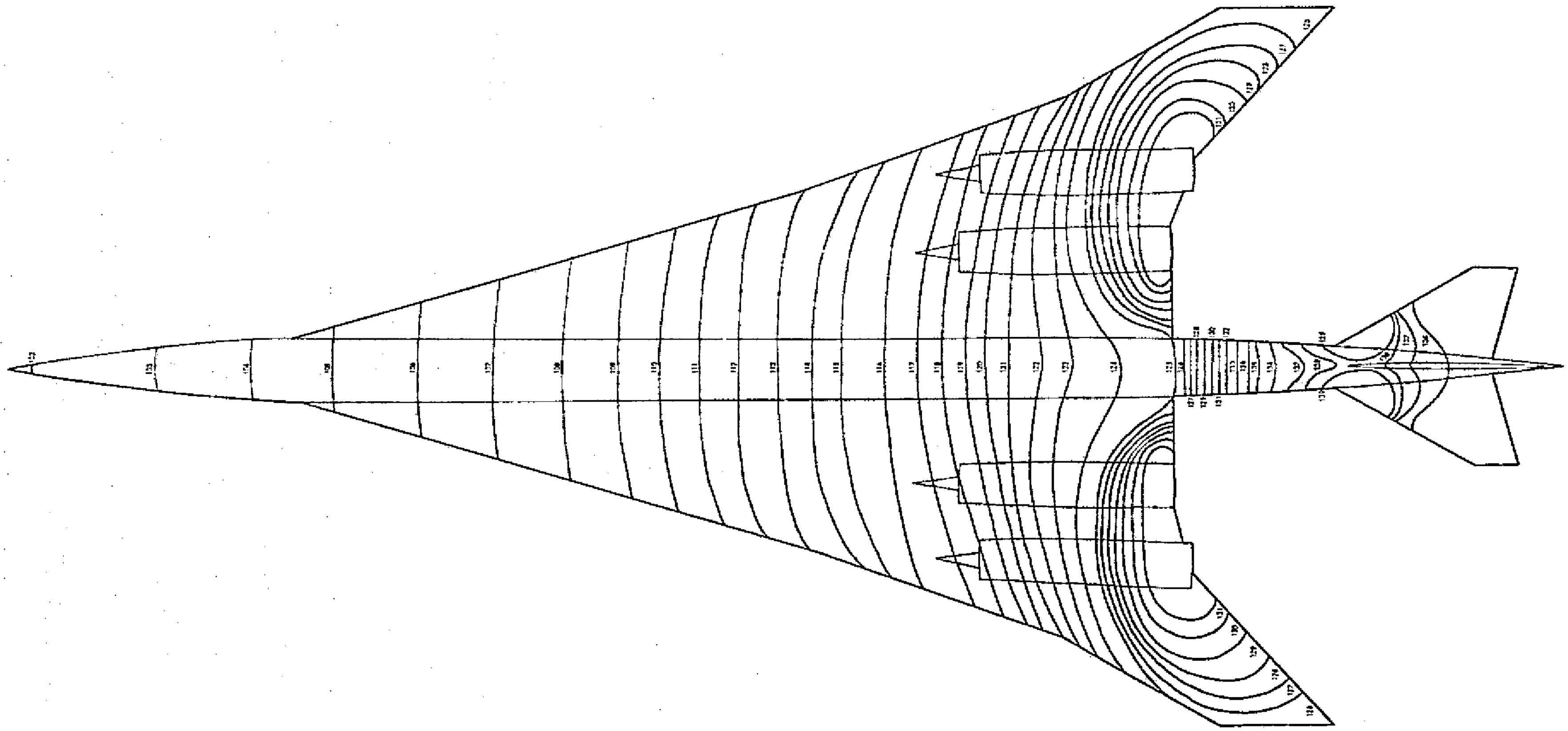


Figure 14-13. 125 Hz. Octave Band Pressure Levels - Forward Mounted Engine

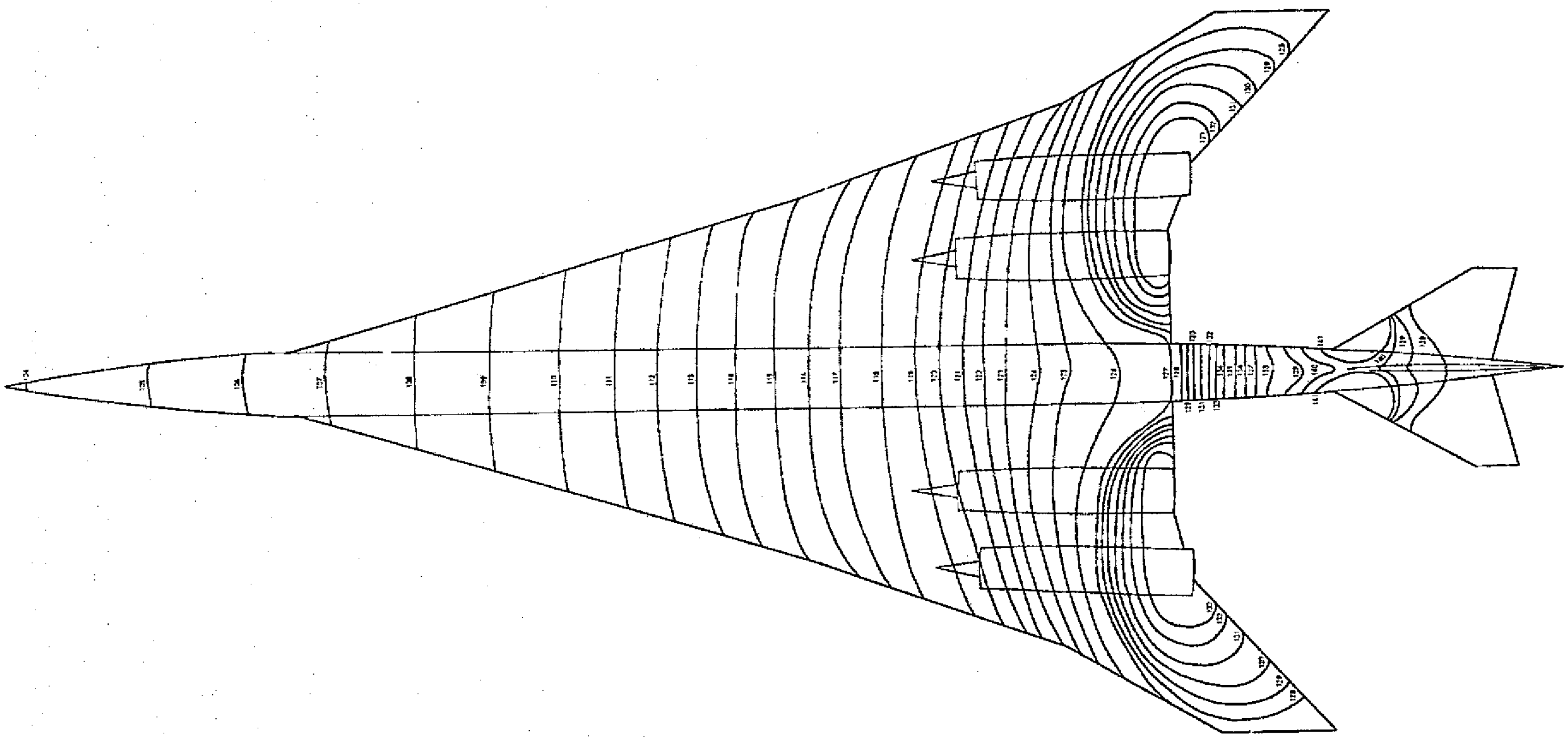


Figure 14-14. 250 Hz. Octave Band Pressure Levels - Forward Mounted Engine

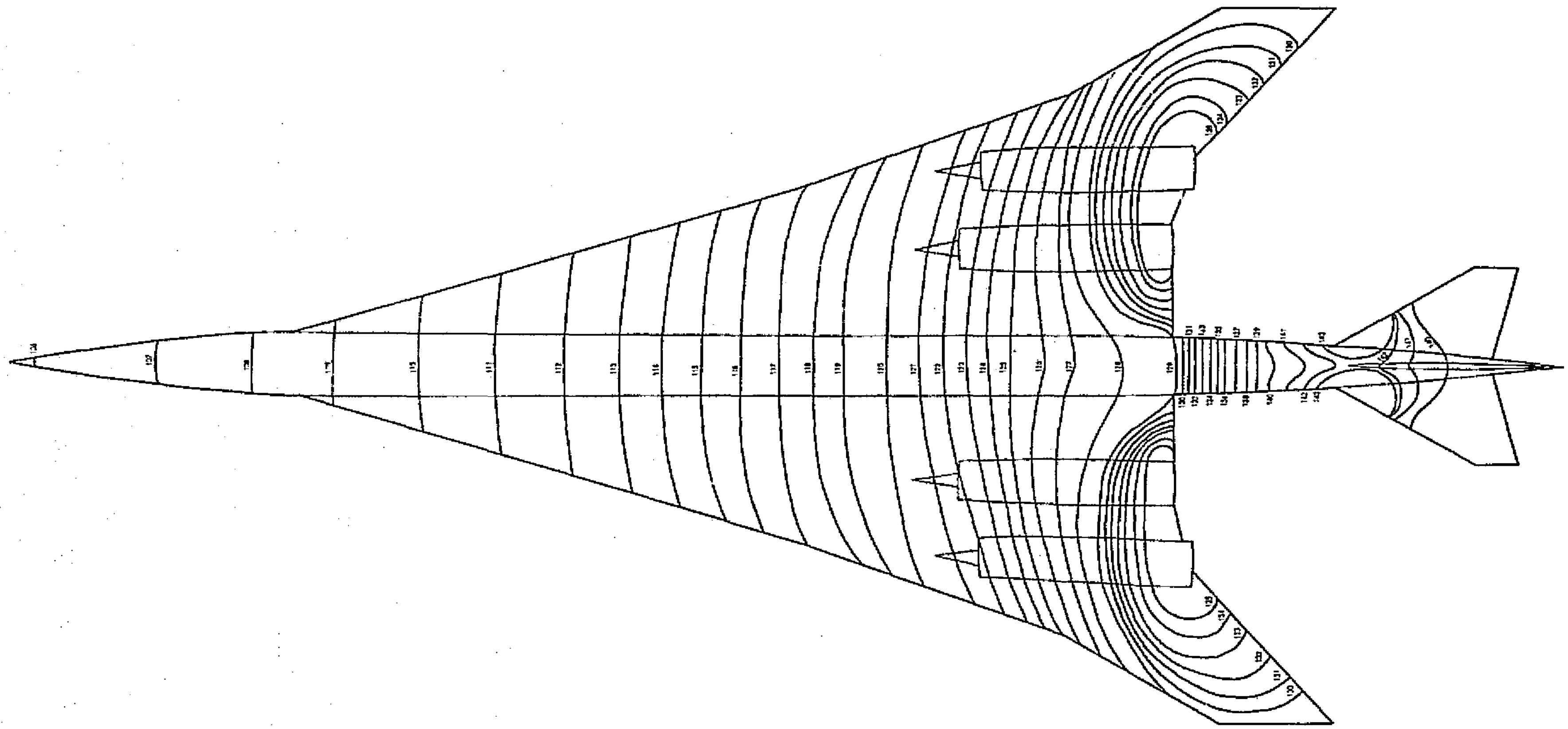


Figure 14-15. 500 Hz. Octave Band Pressure Levels - Forward Mounted Engine

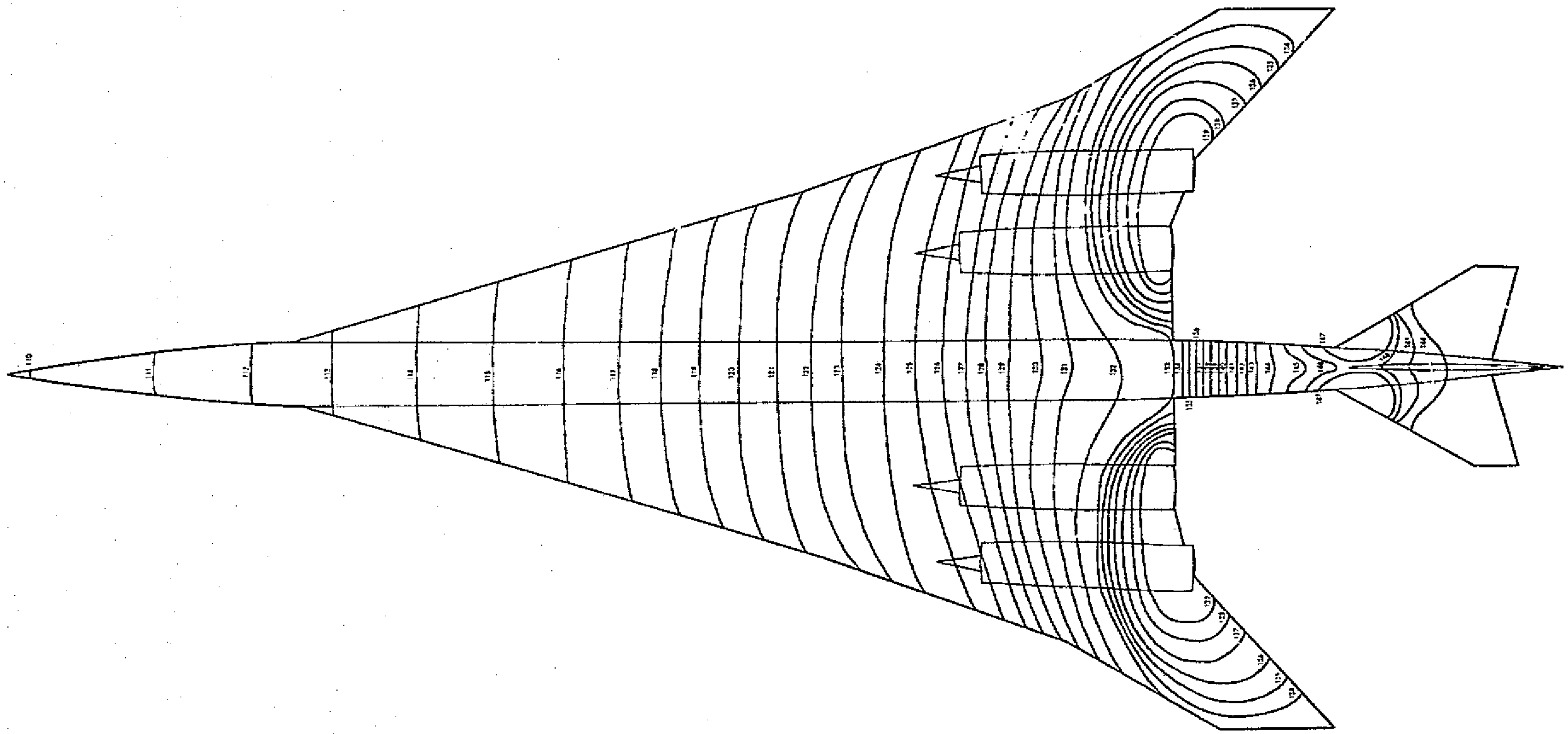


Figure 14-16. 1000 Hz. Octave Band Pressure Levels - Forward Mounted Engine

For structure which must withstand broadband random noise, the resonant condition cannot be avoided. Therefore adequate stiffness must be designed into the structure to keep the acoustically induced stresses sufficiently low to avoid fatigue cracking.

The amplitude of vibratory response which the structure can withstand for a satisfactory period of time without fatigue cracking is highly sensitive to the "quality of detail design", the second property mentioned above. This is a consequence of the fact that it is not the average or "nominal" value but the "maximum" value of the vibratory stresses which limits the fatigue life of a structure. Therefore, it is important to give careful attention to the details of design in order to avoid high concentrations of stresses in localized areas.

Because of the dependency of fatigue resistance on the quality of detail design, it is not possible to predict the fatigue life of a panel by analysis alone. Therefore design charts were used which are based on the analysis of the response of structure to broadband random excitation for which fatigue allowables are chosen to be consistent with fatigue test data for typical aircraft structure. These charts were determined by the analytical and empirical approaches of References 3 and 4.

For the analysis, design charts were used for three different types of panels; they were:

- Orthtropic panels which have unequal stiffness properties along the two principal axes, e.g., convex-beaded and hat-stiffened wing panel concepts.
- Monocoque panels which exhibit appreciable stiffness in both axes and plate theory is applicable, e.g., honeycomb sandwich panels.
- Unstiffened skin panels for analysis of the skin vibrating as a plate between stiffeners.

The series of design charts used to analyze the orthotropic panel concepts were obtained from Reference 3 and are shown in Figure 14-17. This figure outlines the design charts used for determining the allowable spectrum level and natural frequency of the panel, Figure 14-17a and 14-17b, respectively. In addition for completeness, the applied sound spectrum level is shown in Figure 14-17c.

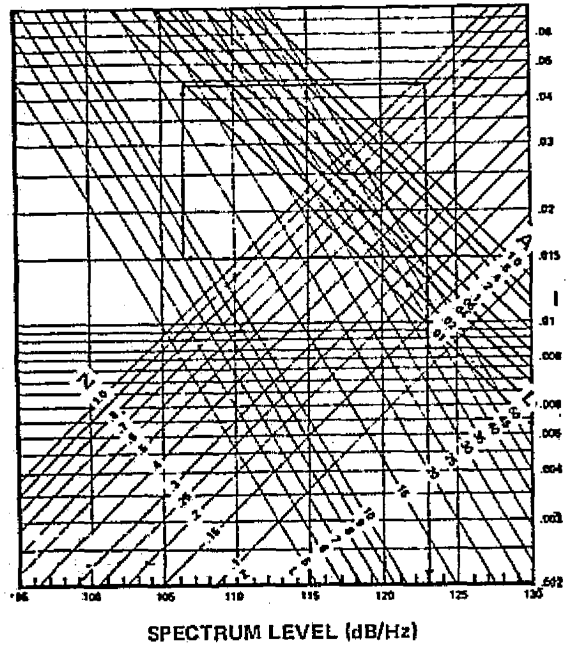


FIGURE 14-17a
ALLOWABLE SOUND
SPECTRUM LEVEL

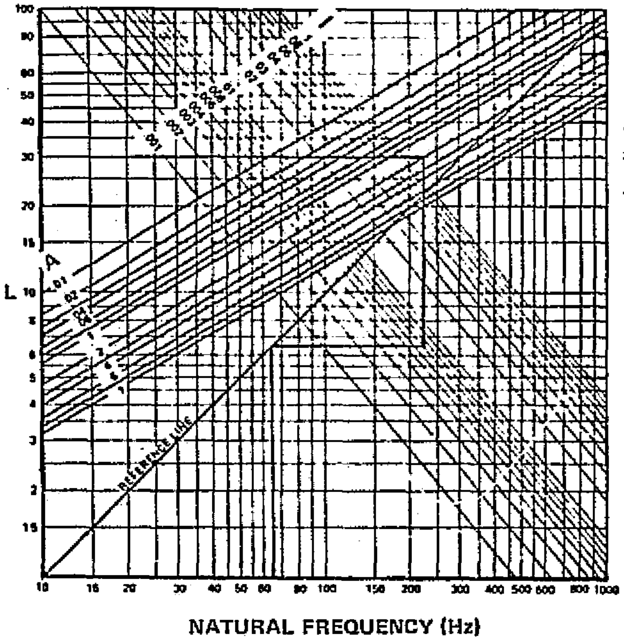


FIGURE 14-17b
NATURAL FREQUENCY

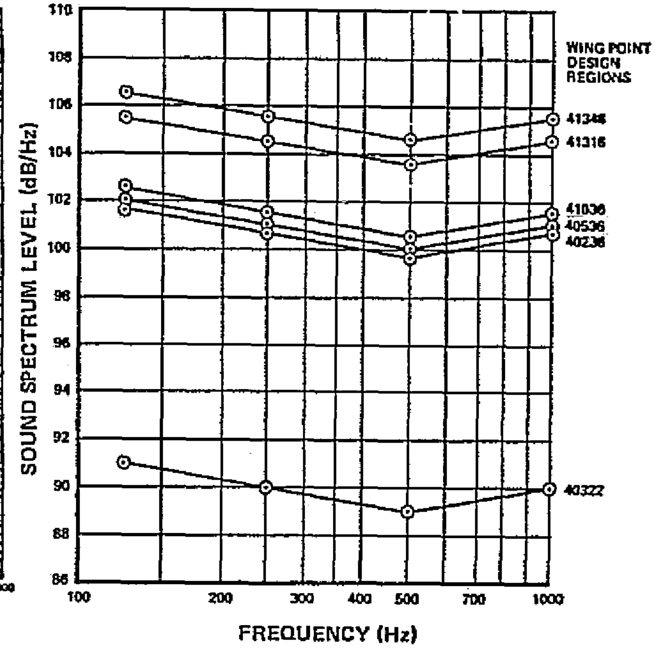


FIGURE 14-17c
ENVIRONMENT SPECT. LEVEL
WING POINT DESIGN REGIONS

- PANEL ALLOWABLE SOUND SPECTRUM LEVEL (PER FIGURE 14-17a)
  ALLOWABLE dB/Hz = FUNCTION (I,A,L,Z)
- PANEL NATURAL FREQUENCY (PER FIGURE 14-17b)
  NATURAL FREQ (Hz) = FUNCTION (I,A,L)
- ENVIRONMENT SOUND SPECTRUM LEVEL (PER FIGURE 14-17c)
- SONIC FATIGUE MARGIN (ALLOWABLE SOUND SPECTRUM LEVEL – ENVIRONMENT SOUND SPECTRUM LEVEL)

Figure 14-17. Orthotropic Panel Sonic Fatigue Design Chart

The monocoque concepts, honeycomb sandwich wing panels were analyzed for both face sheet and edge failure modes. Design charts based on the empirical equations presented in Reference 4 were used in the analysis of these concepts. A sample design chart for determining the face sheet allowable sound spectrum level is shown in Figure 14-18. The corresponding honeycomb sandwich design chart for the edge capability is presented in Figure 83 of Reference 4. The fundamental frequency was calculated using the method presented in the above reference and is shown in Figure 14-19.

The skin panel charts of Figure 14-20 outline the method used to determine the capability of the skin between stiffeners. The natural frequency and allowable sound spectrum level are found from Figures 14-20a and 14-20b, respectively. The applied sound spectrum level is included as Figure 14-20c.

Sonic fatigue analyses require that the applied acoustic environment be defined in terms of sound spectrum levels (db/Hz). The sound spectrum levels are a measure of the acoustic energy contained in a one Hertz bandwidth centered at a specified frequency. The sound spectrum levels were defined for the wing and fuselage point design regions for frequencies of 63, 125, 250, 500, and 1000 Hz. Figures 14-21 and 14-22 present the wing and fuselage sound spectrum levels, respectively. A smooth curve was constructed through the sound spectrum levels at these frequencies to approximate the spectral distribution of the acoustic environment.

The above sound spectrum levels were determined for the baseline airplane by reducing the octave band noise level contours presented in Figures 14-7 through 14-11 to one Hertz bands. This was accomplished by subtracting the following values from the contour levels given in these figures.

Frequency (Hz)	ΔdB
63	16.5
125	19.5
250	22.5
500	25.5
1000	28.5

As a relative merit of each structural concept a sonic fatigue margin was calculated. This margin, allowable panel sound spectrum level minus the applied sound spectrum level, allows the reader to numerically assess the capability of each concept.

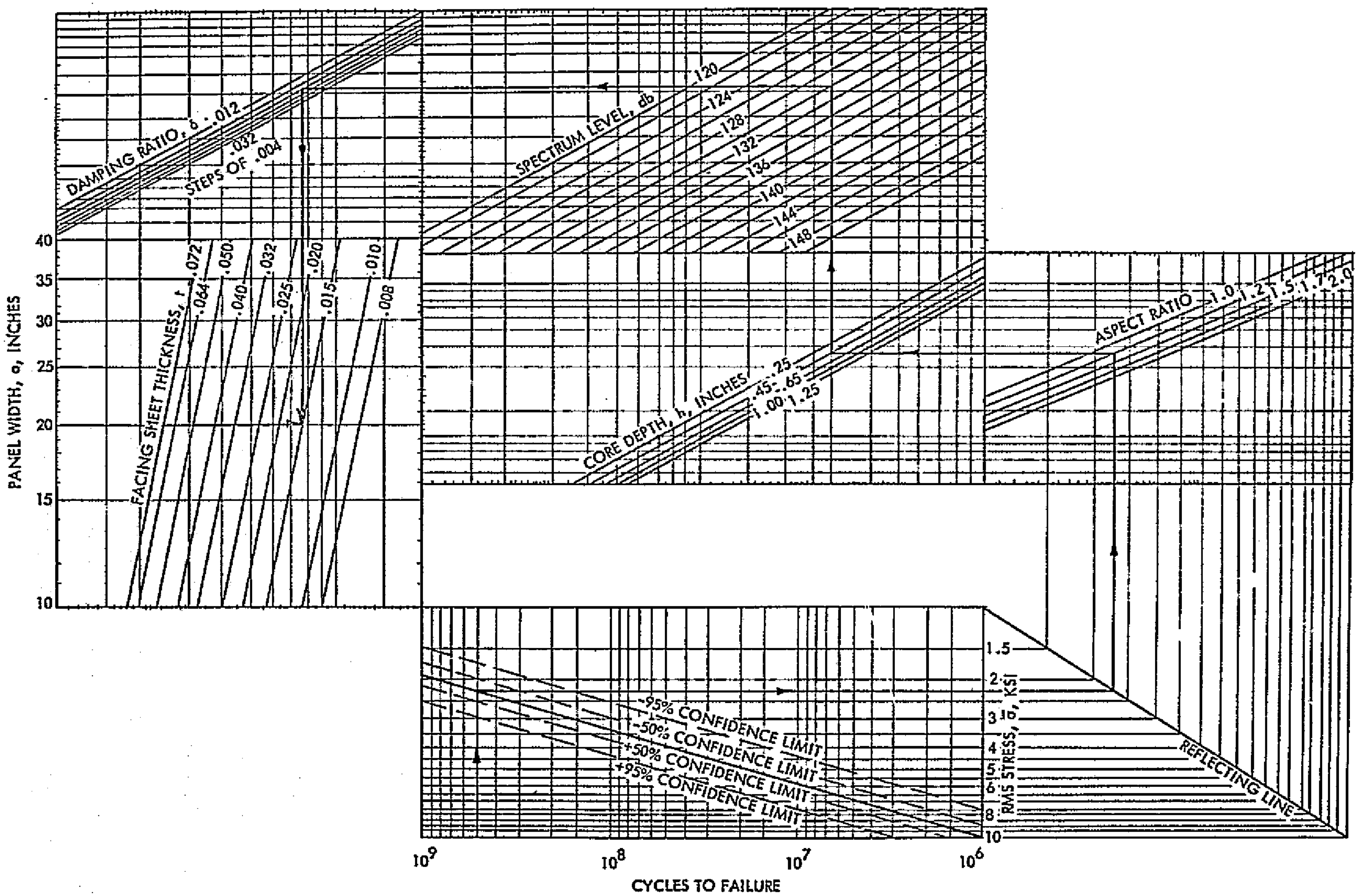


Figure 14-18. Honeycomb Sandwich Panel Sonic Fatigue Design Chart

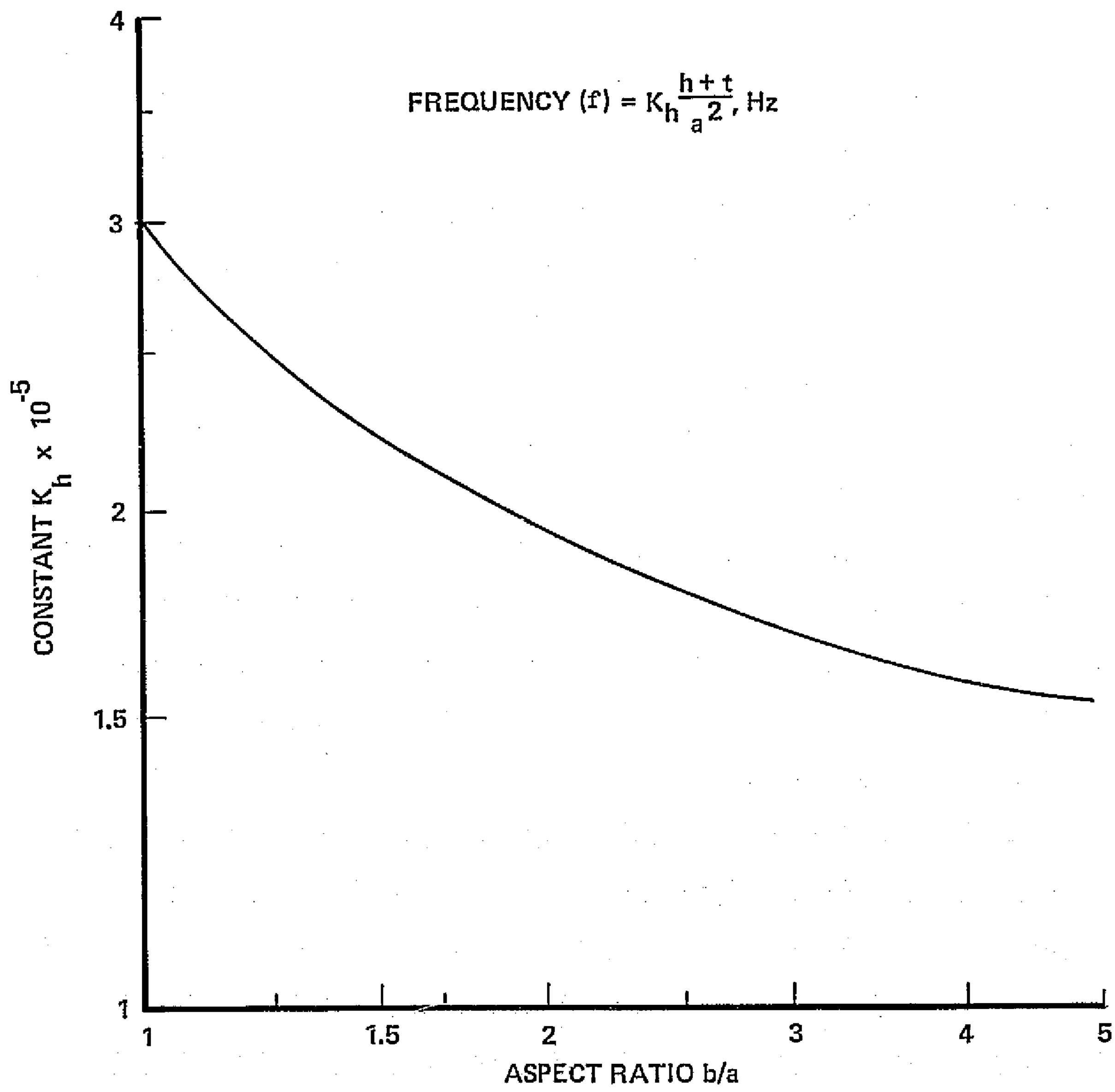


Figure 14-19. Fundamental Frequency of Honeycomb Sandwich Panel

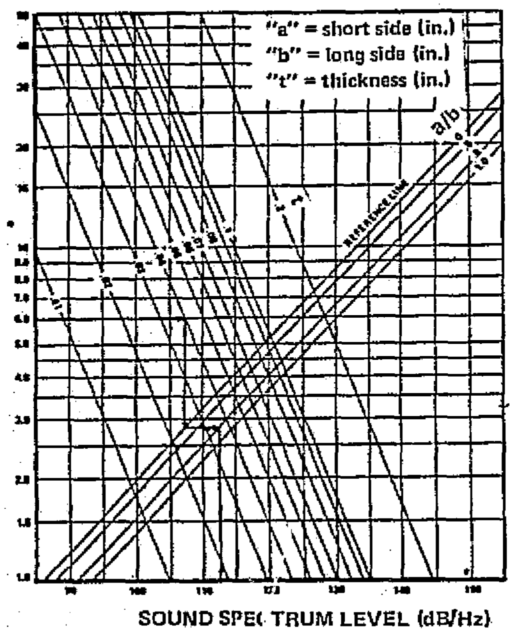


FIGURE 14-20a
ALLOWABLE SOUND
SPECTRUM LEVEL

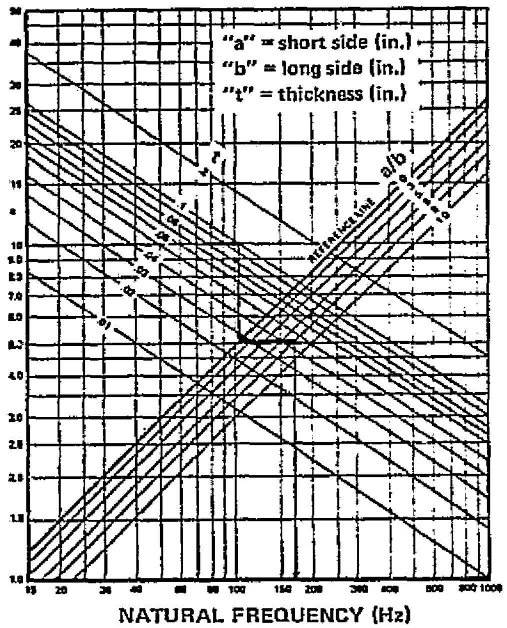


FIGURE 14-20b
NATURAL FREQUENCY

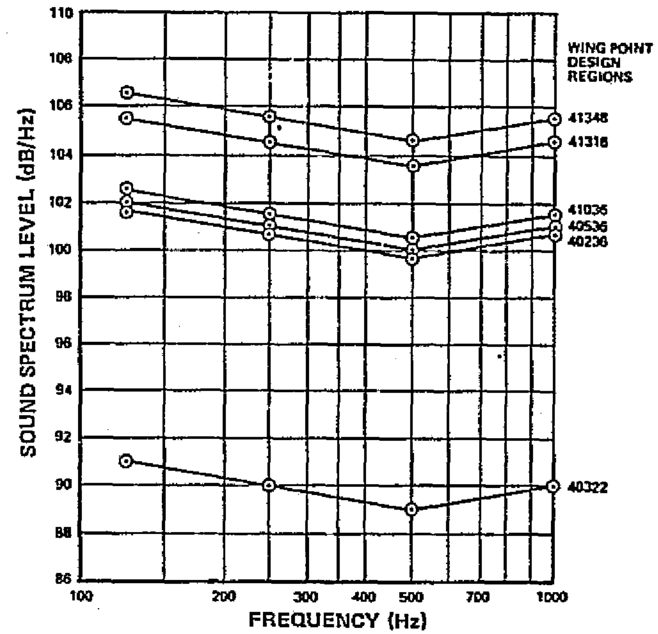


FIGURE 14-20c
ENVIRONMENT SOUND SPECTRUM
LEVEL WING POINT DESIGN REGIONS

- SKIN ALLOWABLE SOUND SPECTRUM LEVEL (PER FIGURE 14-20a)
  ALLOWABLE dB/Hz = FUNCTION (t,a,a/b)
- SKIN NATURAL FREQUENCY (PER FIGURE 14-20b)
  NATURAL FREQUENCY (f) = FUNCTION (t,a,a/b)
- ENVIRONMENT SOUND SPECTRUM LEVEL (PER FIGURE 14-20c)
- SONIC FATIGUE MARGIN (ALLOWABLE dB/Hz – ENVIRONMENT dB/Hz)

Figure 14-20. Skin Panel Sonic Fatigue Design Chart

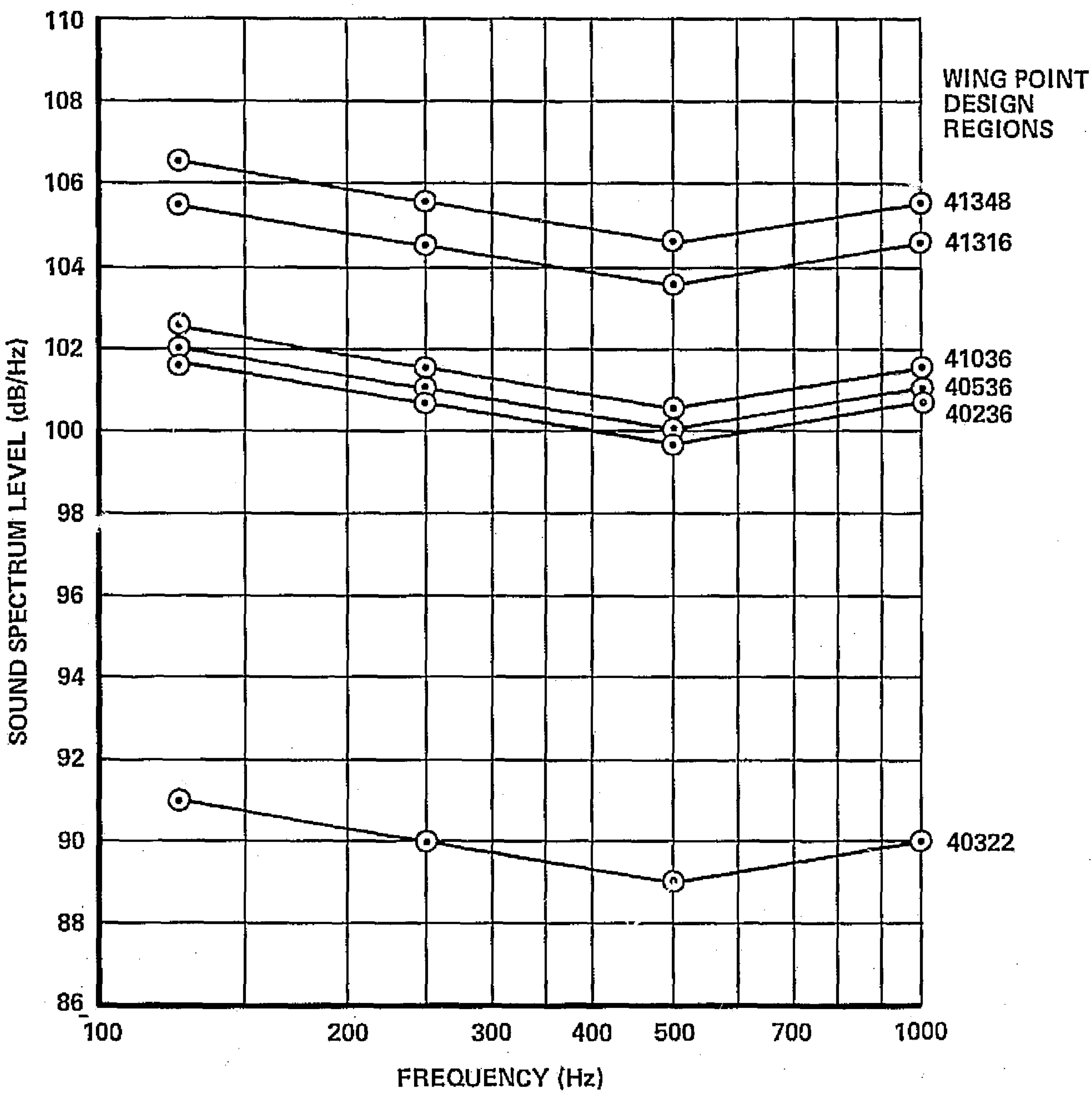


Figure 14-21. Wing Point Design Sound Spectrum Levels, Baseline Engine Location

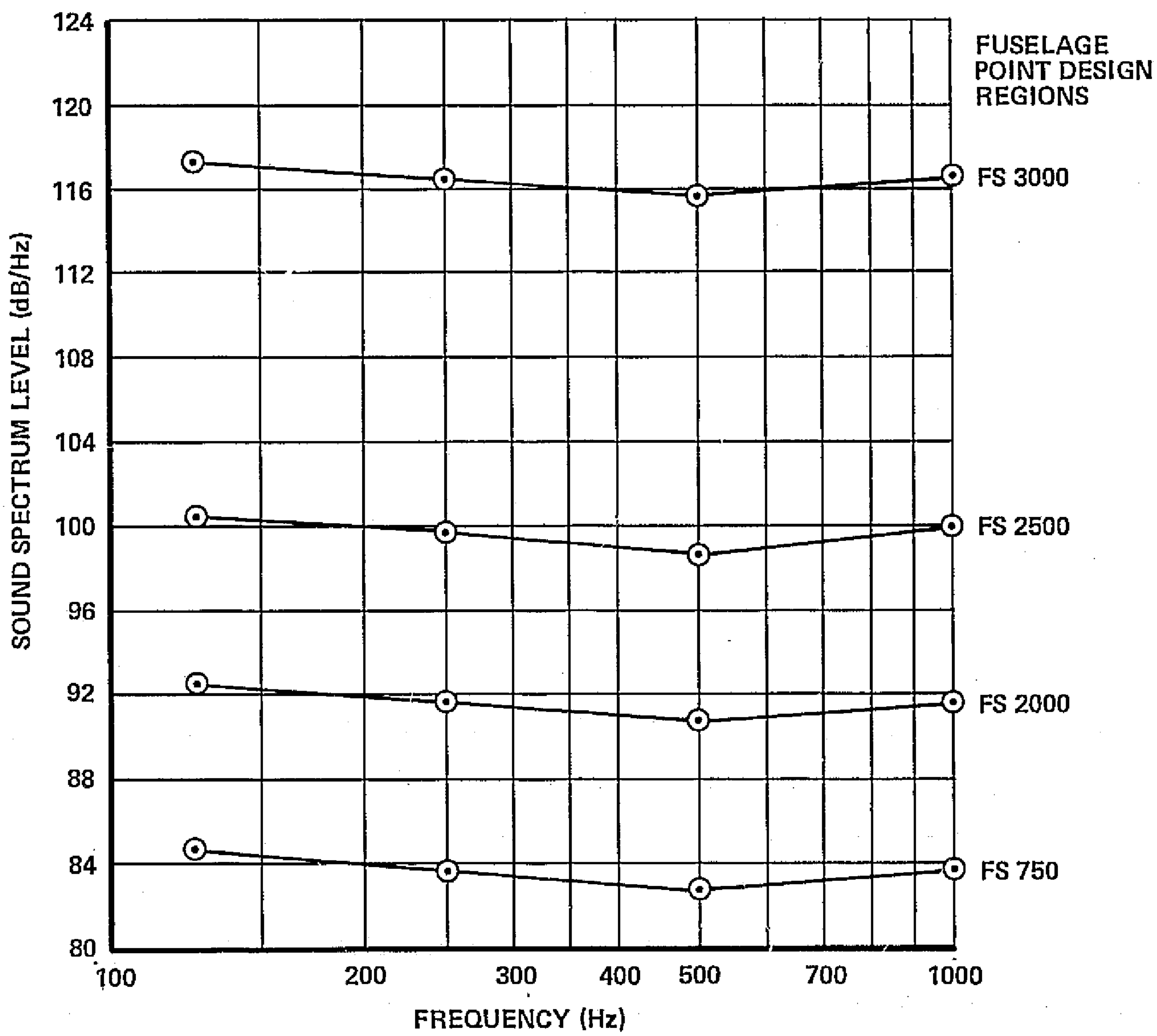


Figure 14-22. Fuselage Point Design Sound Spectrum Levels, Baseline Engine Location

SONIC FATIGUE ANALYSIS - TASK I

In conjunction with the Task I analytical studies, the most promising wing and fuselage structural candidates surviving the initial screening were subjected to sonic fatigue evaluation. The wing candidates evaluated were the least weight concept representative of each of the three general types of load carrying structure, i.e., chordwise, spanwise, and monocoque. Similarily, the fuselage arrangement analyzed represented the combination of structural concept which afforded the minimum weight fuselage design.

Wing Analysis

Sonic fatigue analyses are conducted on each of the wing concepts at the six point design regions. The upper and lower surface panels were analyzed at each point design region. Figure 14-23 presents these point design regions overlayed on the structural model planform.

The general types of wing structure and the most promising surface panel concept for each type were:

- Chordwise - Circular arc convex-beaded concept.
- Spanwise - Hat-stiffened concept.
- Monocoque - Honeycomb sandwich concept.

The panel cross-sectional properties for the convex-beaded concept (chordwise arrangement) are shown in Tables 14-3 and 14-4. These data reflect the results of the strength analysis conducted to define the minimum weight design and the associated spar spacing. For the minimum weight chordwise design, convex-beaded concept, a spar spacing of approximately 20 inches resulted in the least weight design. The sonic fatigue capability of the surface panels for this configuration were evaluated at the six point design regions.

The surface panel geometry for the least weight spanwise arrangement, hat-stiffened concept, is shown in Tables 14-5 and 14-6. The minimum weight rib spacing for this concept is approximately 30 inches.

Similarly, the panel geometry for the honeycomb sandwich concept, least weight monocoque concept, is shown in Tables 14-7 and 14-8. The minimum weight panel dimensions are 20 inches by 60 inches, spar and rib spacing respectively.

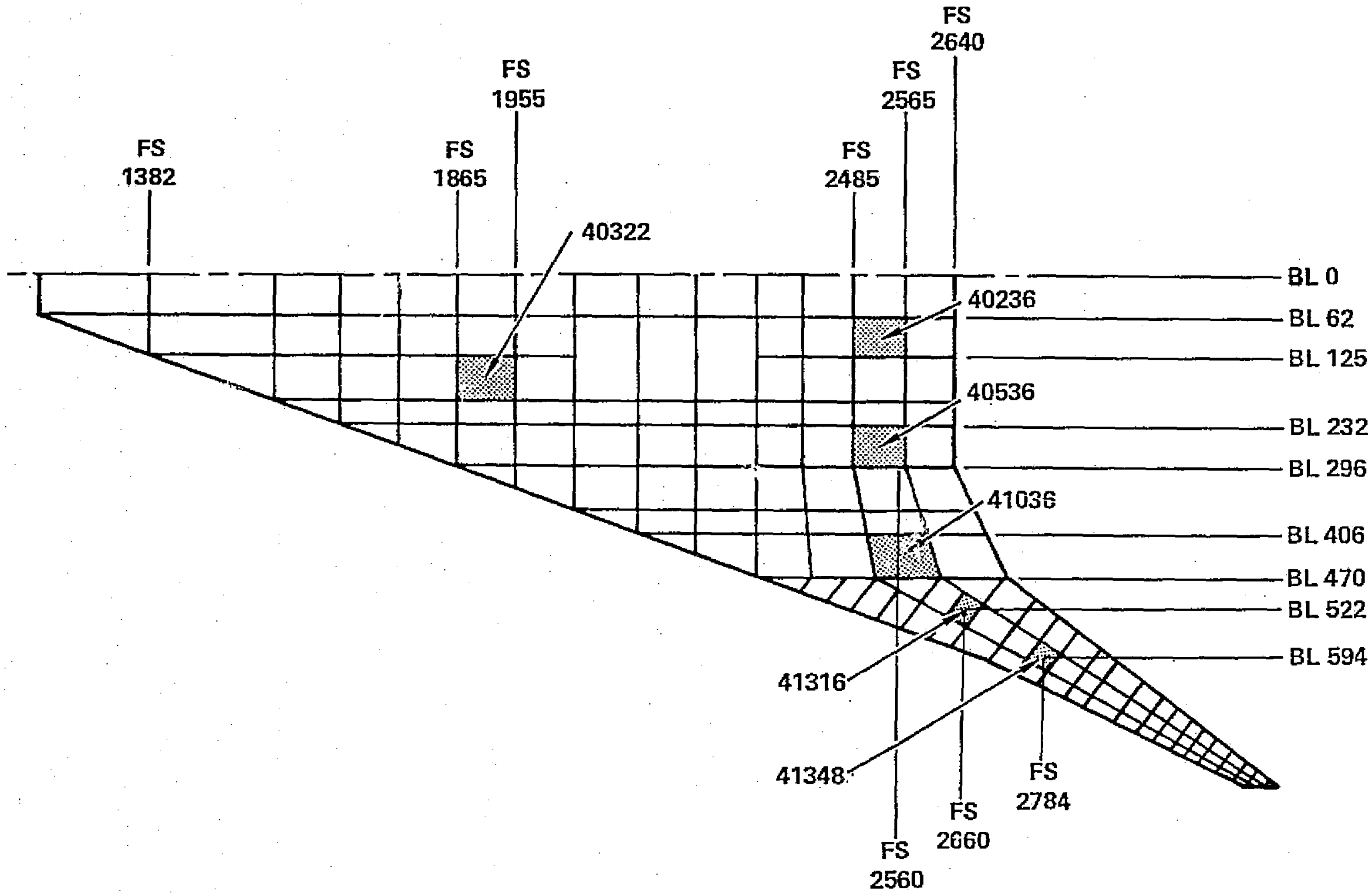
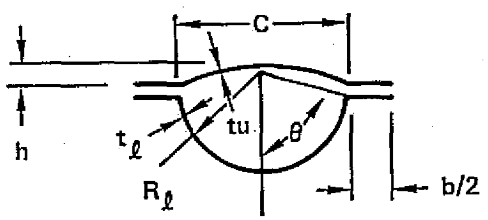


Figure 14-23. Definition of Wing Point Design Regions

TABLE 14-3. WING PANEL GEOMETRY, TASK I CHORDWISE ARRANGEMENT - CONVEX BEADED CONCEPT

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_ℓ	(in)	.015	.021	.031	.015	.020	.025	.025	.035	.040	.024	.028	.033	.025	.033	.038	.023	.020	.023
t_u	(in)	.015	.025	.026	.020	.020	.025	.035	.036	.040	.025	.029	.037	.036	.037	.041	.028	.030	.038
R_ℓ	(in)	0.9	1.2	1.4	0.9	1.4	1.8	0.9	1.1	1.4	0.8	1.0	1.4	0.9	1.1	1.4	0.7	0.8	0.9
θ	(deg)	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87
b	(in)	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75
MASS DATA:																			
$\overline{t}$	(in)	.036	.055	.070	.041	.049	.061	.070	.085	.097	.058	.068	.084	.071	.084	.095	.059	.058	.070
w	(lb/ft^2)	.0825	1.263	1.619	0.942	1.120	1.413	1.609	1.965	2.241	1.335	1.570	1.943	1.632	1.925	2.199	1.366	1.328	1.616
CRITICAL CONDITION		20	20	20	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31

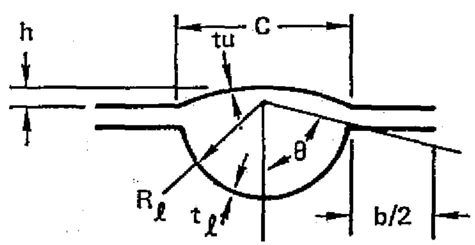


PANEL CONCEPT:

CIRCULAR ARC–CONVEX BEADED SKIN (h/c= 0.10)

TABLE 14-4. WING PANEL GEOMETRY, TASK I CHORDWISE ARRANGEMENT – CONVEX BEADED CONCEPT

POINT DESIGN REGION		40236						41036						41316					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02	.51	.76	1.02
	(in)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	[illegible]	20	30	40
DIMENSIONS:																			
t_ℓ	(in)	.019	.024	.030	.022	.028	.034	.024	.028	.033	.020	.021	.023	.028	.030	.038	.026	.033	.037
t_u	(in)	.019	.024	.028	.025	.030	.033	.030	.037	.040	.030	.029	.030	.072	.072	.067	.051	.046	.050
R_ℓ	(in)	0.7	1.0	1.3	0.9	1.2	1.5	0.8	1.1	1.3	0.7	0.8	1.0	0.9	1.1	1.4	0.8	1.0	1.2
θ	(deg)	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87	87
b	(in)	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75	.75
MASS DATA:																			
$\bar{t}$	(in)	.045	.058	.071	.055	.070	.082	.063	.077	.087	.057	.058	.062	.112	.115	.122	.087	.092	.103
w	(1b/ft^2)	1.032	1.325	1.629	1.279	1.606	1.887	1.452	1.766	2.007	1.320	1.336	1.435	2.571	2.650	2.811	2.007	2.129	2.366
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



PANEL CONCEPT:

CIRCULAR ARC–CONVEX BEADED SKIN (h/c= 0.10)

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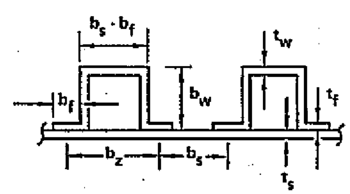
TABLE 14-5. WING PANEL GEOMETRY, TASK I SPANWISE ARRANGEMENT-FLAT STIFFENED CONCEPT

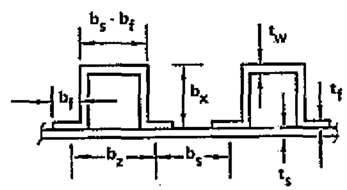
POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.0660	0.0890	0.1110	0.0520	0.0520	0.0530	0.2260	0.2390	0.2590	0.2770	0.2640	0.2670	0.1480	0.1710	0.2010	0.1600	0.1610	0.1610
	(in.)	0.0260	0.0350	0.0440	0.0200	0.0200	0.0210	0.0890	0.0940	0.1020	0.1090	0.1040	0.1050	0.0580	0.0672	0.0792	0.0630	0.0630	0.0640
$b_s = b_w = b_z$	(cm)	2.2800	3.2400	4.1700	2.0100	2.4600	2.8800	4.4600	5.3500	6.3700	5.1100	5.6900	6.4700	3.4100	4.4800	5.6100	3.5700	4.3400	5.0200
	(in.)	0.8960	1.2750	1.6420	0.7910	0.9690	1.1330	1.7570	2.1070	2.5090	2.0120	2.2410	2.5490	1.3440	1.7630	2.2090	1.4050	1.7110	1.9790
$t_w = t_f$	(cm)	0.0610	0.0830	0.1030	0.0480	0.0480	0.0490	0.2090	0.2210	0.2390	0.2550	0.2440	0.2460	0.1360	0.1570	0.1860	0.1470	0.1480	0.1490
	(in.)	0.0240	0.0320	0.0400	0.0190	0.0190	0.0190	0.0820	0.0870	0.0940	0.1000	0.0960	0.0970	0.0540	0.0620	0.0730	0.0580	0.0580	0.0590
b_f	(cm)	0.6830	0.9730	1.2500	0.6020	0.7390	0.8640	1.3400	1.6000	1.9100	1.5200	1.7100	1.9400	1.0200	1.3400	1.6800	1.0700	1.3000	1.5100
	(in.)	0.2690	0.3830	0.4930	0.2370	0.2910	0.3400	0.5270	0.6320	0.7530	0.6000	0.6720	0.7650	0.4030	0.5290	0.6630	0.4210	0.5130	0.5940
$b_s \cdot b_f$	(cm)	1.5900	2.2700	2.9200	1.4100	1.7200	2.0100	3.1200	3.7500	4.4600	3.5600	3.9800	4.5300	2.3900	3.1300	3.9300	2.5000	3.0400	3.5200
	(in.)	0.6270	0.8930	1.1500	0.5540	0.6780	0.7930	1.2300	1.4750	1.7570	1.4010	1.5690	1.7840	0.9410	1.2340	1.5470	0.9830	1.1980	1.3860
MASS DATA:																			
$\bar{t}$	(cm)	0.1669	0.2253	0.2804	0.1302	0.1302	0.1335	0.5716	0.6038	0.6544	0.6981	0.6665	0.6749	0.3735	0.4306	0.5074	0.4025	0.4056	0.4072
	(in.)	0.0657	0.0887	0.1104	0.0512	0.0512	0.0526	0.2250	0.2377	0.2577	0.2748	0.2624	0.2657	0.1470	0.1695	0.1998	0.1584	0.1597	0.1603
w	(kg/m^2)	7.3900	9.9800	12.4200	5.7700	5.7700	5.9100	25.3200	26.7400	28.9900	30.9000	29.5200	29.8900	16.5400	19.0700	22.4700	17.8200	17.9600	18.0400
	(lb/ft^2)	1.5140	2.0440	2.5430	1.1810	1.1810	1.2110	5.1850	5.4760	5.9370	6.3300	6.0460	6.1220	3.3880	3.9000	4.6020	3.6510	3.6790	3.6940
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



HAT SECTION STIFFEN

TABLE 14-6. WING PANEL GEOMETRY, TASK I SPANWISE ARRANGEMENT - FLAT STIFFENED CONCEPT

POINT DESIGN REGIONS		40236						41036						41316					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
RIB SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
DIMENSIONS:																			
t_s	(cm)	0.2090	0.2290	0.2650	0.2260	0.2250	0.2300	0.1600	0.1780	0.1960	0.1850	0.1810	0.1800	0.2490	0.2540	0.2730	0.2760	0.2770	0.2730
	(in.)	0.0820	0.0900	0.1040	0.0890	0.0890	0.0910	0.0630	0.0700	0.0770	0.0730	0.0710	0.0710	0.0980	0.1000	0.1080	0.1090	0.1090	0.1080
$b_s = b_w = b_z$	(cm)	4.2400	5.2200	6.4400	4.4600	5.1700	6.0000	3.5700	4.5800	5.5400	3.9100	4.6000	5.3100	4.7400	5.5500	6.5400	5.0800	5.8600	6.5400
	(in.)	1.6680	2.0540	2.5360	1.7570	2.0340	2.3640	1.4050	1.8030	2.1800	1.5400	1.8130	2.0900	1.8670	2.1860	2.5750	2.0020	2.3090	2.5750
$t_w = t_f$	(cm)	0.1930	0.2120	0.2440	0.2090	0.2080	0.2150	0.1470	0.1650	0.1810	0.1700	0.1670	0.1660	0.2300	0.2340	0.2520	0.2550	0.2560	0.2520
	(in.)	0.0760	0.0830	0.0960	0.0870	0.0820	0.0840	0.0580	0.0650	0.0710	0.0670	0.0660	0.0650	0.0900	0.0920	0.0990	0.1000	0.1010	0.0990
b_f	(cm)	1.2700	1.5600	1.9300	1.3400	1.5500	1.8000	1.0700	1.3700	1.6600	1.1700	1.3800	1.5900	1.4200	1.6700	1.9600	1.5200	1.7600	1.9600
	(in.)	0.5000	0.6160	0.7610	0.5270	0.6100	0.7090	0.4210	0.5410	0.6540	0.4620	0.5440	0.6270	0.5610	0.6560	0.7730	0.6000	0.6930	0.7730
$b_s - b_f$	(cm)	2.9700	3.6500	4.5100	3.1200	3.6200	4.2000	2.5000	3.2000	3.8800	2.7400	3.2200	3.7200	3.3200	3.8900	4.5800	3.5600	4.1100	4.5800
	(in.)	1.1680	1.4380	1.7750	1.2300	1.4240	1.6550	0.9830	1.2620	1.5260	1.0780	1.2690	1.4630	1.3070	1.5300	1.8030	1.4010	1.6170	1.8030
MASS DATA:																			
$\overline{t}$	(cm)	0.5277	0.5786	0.7389	0.5716	0.5688	0.5808	0.4025	0.4506	0.4940	0.4659	0.4556	0.4539	0.6273	0.6409	0.6889	0.6981	0.6988	0.6889
	(in.)	0.2078	0.2278	0.2941	0.2250	0.2239	0.2287	0.1584	0.1774	0.1945	0.1834	0.1794	0.1787	0.2470	0.2523	0.2712	0.2748	0.2751	0.2712
w	(kg/m^2)	23.3700	25.6200	33.0800	25.3200	25.1900	25.7200	17.8200	19.9600	21.8800	20.6300	20.1800	20.1100	27.7800	28.3900	30.5100	30.9000	30.9400	30.5100
	(lb/ft^2)	4.7900	5.2500	6.7800	5.1800	5.1600	5.2700	3.6500	4.0900	4.4800	4.2300	4.1300	4.1200	5.6900	5.8100	6.2500	6.3300	6.3400	6.2500
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31



HAT SECTION STIFFENED

14-36

TABLE 14-7. WING PANEL GEOMETRY, TASK I MONOCOQUE ARRANGEMENT - HONEYCOMB SANDWICH CONCEPT

POINT DESIGN REGION		40322						40536						41348					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
RIB SPACING	(m)	3.30	3.30	3.30	3.30	3.30	3.30	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52
	(in.)	130	130	130	130	130	130	60	60	60	60	60	60	60	60	60	60	60	60
ASPECT RATIO		0.15	0.23	0.31	0.15	0.23	0.31	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67
DIMENSIONS:																			
H	(cm)	1.577	2.530	3.713	2.101	2.652	5.184	2.126	3.233	3.762	0.737	1.153	1.984	1.826	2.647	3.279	0.561	0.917	1.156
	(in.)	0.621	0.996	1.462	0.827	1.044	2.041	0.837	1.273	1.481	0.290	0.454	0.781	0.719	1.042	1.291	0.221	0.361	0.455
t_1	(cm)	0.038	0.046	0.058	0.028	0.038	0.038	0.135	0.132	0.127	0.193	0.193	0.221	0.089	0.091	0.099	0.119	0.178	0.137
	(in.)	0.015	0.018	0.023	0.011	0.015	0.015	0.053	0.052	0.050	0.076	0.076	0.087	0.035	0.036	0.039	0.047	0.070	0.054
t_2	(cm)	0.038	0.038	0.038	0.051	0.051	0.051	0.132	0.130	0.127	0.155	0.160	0.135	0.097	0.097	0.102	0.112	0.051	0.099
	(in.)	0.015	0.015	0.015	0.020	0.020	0.020	0.052	0.051	0.050	0.061	0.063	0.053	0.038	0.038	0.040	0.044	0.020	0.039
t_c	(cm)	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005
	(in.)	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002
S	(cm)	0.597	0.716	0.795	1.270	1.270	1.270	0.655	0.470	0.424	1.270	1.270	1.270	0.808	0.655	0.737	1.270	1.270	1.270
	(in.)	0.235	0.282	0.313	0.500	0.500	0.500	0.258	0.185	0.167	0.500	0.500	0.500	0.318	0.258	0.290	0.500	0.500	0.500
MASS DATA:																			
$\overline{t}$	(cm)	0.102	0.119	0.142	0.097	0.109	0.130	0.297	0.333	0.353	0.353	0.361	0.368	0.208	0.226	0.244	0.234	0.234	0.241
	(in.)	0.040	0.047	0.056	0.038	0.043	0.051	0.117	0.131	0.139	0.139	0.142	0.145	0.082	0.089	0.096	0.092	0.092	0.095
W	$(kg \cdot m^{-2})$	4.511	5.263	6.289	4.248	4.863	5.737	13.124	14.740	15.682	15.634	16.000	16.346	9.169	10.009	10.785	10.336	10.370	10.658
	$lb \cdot ft^{-2})$	0.924	1.078	1.288	0.870	0.996	1.175	2.688	3.019	3.212	3.202	3.277	3.348	1.878	2.050	2.209	2.117	2.124	2.183
W_c	$(kg \cdot m^{-2})$	1.040	1.538	2.051	0.718	0.908	1.850	1.279	3.095	4.458	0.137	0.283	0.576	0.913	1.694	1.880	0.117	0.244	0.327
	$(lb \cdot ft^{-2})$	0.213	0.315	0.420	0.147	0.186	0.379	0.262	0.634	0.913	0.028	0.058	0.118	0.187	0.347	0.385	0.024	0.050	0.067
ρ_c	$(kg \cdot m^{-3})$	75.431	62.824	56.641	35.433	35.433	35.433	68.880	104.25	127.12	35.433	35.433	35.433	55.632	68.863	61.062	35.433	35.433	35.433
	$(lb \cdot ft^{-3})$	4.709	3.922	3.536	2.212	2.212	2.212	4.300	6.508	7.936	2.212	2.212	2.212	3.473	4.299	3.812	2.212	2.212	2.212
CRITICAL CONDITION		20	20	20	20	20	20	31	31	31	31	31	31	31	31	31	31	31	31

NOTE: (1) ASPECT RATIO = $L_{P,X}/L_{P,Y}$
(2) BRAZE MATERIAL NOT INCLUDED

ORIGINAL PAGE IS OF POOR QUALITY

TABLE 14-8. WING PANEL GEOMETRY, TASK I MONOCOQUE ARRANGEMENT - HONEYCOMB SANDWICH CONCEPT

POINT DESIGN REGION		40236						41036						41316					
SURFACE		UPPER			LOWER			UPPER			LOWER			UPPER			LOWER		
SPAR SPACING	(m)	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02	0.51	0.76	1.02
	(in.)	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40	20	30	40
RIB SPACING	(m)	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52	1.52
	(in.)	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60	60
ASPECT RATIO		0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67	0.33	0.50	0.67
DIMENSIONS:																			
H	(cm)	1.958	2.883	3.818	1.059	1.412	2.360	1.760	2.525	3.254	0.655	0.848	[illegible]	1.834	2.545	3.409	0.488	0.622	0.808
	(in.)	0.771	1.135	1.503	0.417	0.556	0.926	0.693	0.994	1.281	0.258	0.334	0.431	0.722	1.002	1.342	0.192	0.245	0.318
t_1	(cm)	0.119	0.122	0.130	0.234	0.188	0.147	0.084	0.084	0.091	0.119	0.127	0.114	0.132	0.137	0.145	0.178	0.173	0.173
	(in.)	0.047	0.048	0.051	0.092	0.074	0.058	0.033	0.033	0.036	0.047	0.050	0.045	0.052	0.054	0.057	0.070	0.068	0.068
t_2	(cm)	0.124	0.127	0.130	0.069	0.117	0.157	0.084	0.094	0.094	0.091	0.081	0.097	0.135	0.137	0.135	0.167	0.163	0.163
	(in.)	0.049	0.050	0.051	0.027	0.046	0.062	0.033	0.037	0.037	0.036	0.032	0.038	0.053	0.054	0.053	0.062	0.064	0.064
t_c	(cm)	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005	0.005
	(in.)	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002	0.002
S	(cm)	0.947	0.744	0.729	1.270	1.270	1.270	0.747	0.767	0.752	1.270	1.270	1.270	0.925	0.831	0.726	1.270	1.270	1.270
	(in.)	0.373	0.293	0.287	0.500	0.500	0.500	0.294	0.302	0.296	0.500	0.500	0.500	0.364	0.327	0.286	0.500	0.500	0.500
MASS DATA:																			
$\bar{t}$	(cm)	0.264	0.284	0.310	0.310	0.315	0.323	0.191	0.208	0.226	0.213	0.213	0.218	0.284	0.302	0.323	0.335	0.335	0.338
	(in.)	0.104	0.112	0.122	0.122	0.124	0.127	0.075	0.082	0.089	0.084	0.084	0.086	0.112	0.119	0.127	0.132	0.132	0.133
W	$(kg \cdot m^{-2})$	11.674	12.631	13.715	13.720	13.964	14.305	8.393	9.228	9.989	9.443	9.477	9.623	12.597	13.412	14.252	14.857	14.891	15.009
	$(lb \cdot ft^{-2})$	2.391	2.587	2.809	2.810	2.860	2.930	1.719	1.890	2.046	1.934	1.941	1.971	2.580	2.747	2.919	3.043	3.050	3.074
W_c	$(kg \cdot m^{-2})$	0.815	1.592	2.192	0.269	0.391	0.723	0.962	1.377	1.836	0.156	0.225	0.312	0.762	1.230	1.938	0.054	0.103	0.166
	$(lb \cdot ft^{-2})$	0.167	0.326	0.449	0.055	0.080	0.148	0.197	0.282	0.376	0.032	0.046	0.064	0.156	0.252	0.397	0.011	0.021	0.034
ρ_c	$(kg \cdot m^{-3})$	47.687	60.390	61.655	35.433	35.433	35.433	60.277	58.724	59.861	35.433	35.433	35.433	48.616	54.174	61.959	35.433	35.433	35.433
	$(lb \cdot ft^{-3})$	2.977	3.770	3.849	2.212	2.212	2.212	3.763	3.666	3.737	2.212	2.212	2.212	3.036	3.382	3.868	2.212	2.212	2.212
CRITICAL CONDITION		31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31	31

NOTE: (1) ASPECT RATIO = $L_{P,X}/L_{P,Y}$
(2) BRAZE MATERIAL NOT INCLUDED

The exception being the lightly loaded point design region 40322 where a spar/rib spacing of 20 inches and 130 inches resulted in the least weight design.

Chordwise Arrangement - The convex-beaded concept was analyzed using the methods and resulting design charts described in the methods section. A summary of the sonic fatigue results is presented in Table 14-9. With reference to this table, the moment of inertia (I), area (A), and extreme fiber distance (Z) were calculated for these orthotropic panels and used in conjunction with the length to determine the allowable spectrum level, Figure 14-17A. In addition, the applicable frequency chart (Figure 14-17B) was used to define the resonant frequency of each panel. The corresponding environmental spectrum levels for these resonant frequencies were determined from Figure 14-17C or 14-21.

A summary of the sonic fatigue margins, difference between the allowable and environmental spectrum levels, are included in Table 14-9. A minimum margin of +9.4 dB/Hz is noted for the upper surface panel at point design region 41036.

Spanwise Arrangement - The least weight spanwise concept (hat-stiffened) was analyzed similar to the method used for the chordwise arrangement. The panel properties (A, I, and Z) were calculated using the panel geometry defined in Tables 14-5 and 14-6 for the 30 inch rib spacing and 60 inch spar spacing design.

A summary of the spanwise wing panel analyses is presented in Table 14-10. The allowable spectrum level, panel natural frequency, and the applied environmental spectrum level were determined using the same design charts as described for the chordwise analysis. In conclusion, positive sonic fatigue margins exist on the spanwise concept at all point design regions with a minimum margin of +28.1 dB/Hz occuring on the lower surface panel at point design 41348.

Monocoque Arrangement - The honeycomb sandwich panels were analyzed using the methods and resulting design charts described in the methods section. Using these charts the allowable spectrum levels were determined for the panel edge and facing sheets.

Table 14-11 summarizes the results of the Task I honeycomb panel analysis. Included on this table are the pertinent panel properties, natural frequency, allowable spectrum levels, and the applied environmental spectrum level. In addition the sonic fatigue margins are listed and indicated the strength requirements are also adequate for sonic fatigue purposes.

TABLE 14-9. SUMMARY OF WING PANEL SONIC FATIGUE ANALYSES - TASK I CHORDWISE ARRANGEMENT - CONVEX BEADED CONCEPT

POINT DESIGN REGION	WING SURFACE	SPACING (in.)		PANEL PROPERTIES[1]			NATURAL[2] FREQ. (f) (Hz)	SPECTRUM LEVEL[3][4] (dB/Hz)		SONIC[5] FATIGUE MARGIN (dB/Hz)
		a	b	Z (in.)	A (in.^{2}/in.)	I (in.^{4}/in.)		ALLOW.	ENVIR.	
40322	UPPER	20	60	0.644	0.036	0.0018	171.5	115.5	90.5	+25.0
	LOWER	20	60	0.681	0.041	0.0019	167.7	115.9	90.6	+25.3
40236	UPPER	20	60	0.514	0.045	0.0015	142.0	117.0	101.5	+15.5
	LOWER	20	60	0.662	0.056	0.0027	170.1	119.0	101.2	+17.8
40536	UPPER	20	60	0.687	0.070	0.0033	166.8	120.4	101.6	+18.8
	LOWER	20	60	0.583	0.058	0.0024	156.8	119.4	101.8	+17.6
41036	UPPER	20	60	0.604	0.063	0.0025	154.5	111.6	102.2	+9.4
	LOWER	20	60	0.550	0.057	0.0018	136.3	118.0	102.4	+15.6
41316	UPPER	20	60	0.764	0.112	0.0043	151.4	122.3	105.2	+17.1
	LOWER	20	60	0.653	0.087	0.0031	145.5	121.0	105.3	+15.7
41348	UPPER	20	60	0.691	0.071	0.0033	166.3	120.5	106.1	+14.4
	LOWER	20	60	0.530	0.059	0.0019	139.6	118.9	106.4	+12.5

NOTES:

1. PANEL PROPERTIES
   - Z = DISTANCE FROM NEUTRAL AXIS TO EXTREME FIBER, in.
   - A = CROSS-SECTION AREA PER UNIT WIDTH, in.^{2}/in.
   - I = MOMENT OF INERTIA PER UNIT WIDTH, in.^{4}/in.
2. NATURAL FREQUENCY PER FIGURE 14-17B
3. ALLOWABLE SOUND LEVEL PER FIGURE 14-17A
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN = (ALLOWABLE dB/Hz – ENVIRONMENT dB/Hz)

TABLE 14-10. SUMMARY OF WING PANEL SONIC FATIGUE ANALYSES - TASK I SPANWISE ARRANGEMENT - HAT STIFFENED CONCEPT

POINT DESIGN REGION	WING SURFACE	SPACING (in.)		PANEL PROPERTIES(1)			NATURAL(2) FREQ. (f) (Hz)	SPECTRUM LEVEL(3)(4) (dB/Hz)		SONIC(5) FATIGUE MARGIN (dB/Hz)
		a	b	Z (in.)	A (in.^{2}/in.)	I (in.^{4}/in.)		ALLOW.	ENVIR.	
40322	UPPER	60	30	0.894	0.088	0.0216	169.7	127.4	90.6	+36.8
	LOWER	60	30	0.676	0.051	0.0072	129.3	121.5	91.0	+30.5
40236	UPPER	60	30	1.455	0.224	0.1410	272.2	137.5	100.5	+37.0
	LOWER	60	30	1.440	0.221	0.1360	269.6	137.3	100.5	+36.8
40536	UPPER	60	30	1.493	0.234	0.1550	279.2	137.9	100.8	+37.1
	LOWER	60	30	1.590	0.258	0.1930	296.8	139.0	100.7	+38.3
41036	UPPER	60	30	1.273	0.175	0.0852	239.2	134.8	101.6	+33.2
	LOWER	60	30	1.281	0.177	0.0871	240.5	134.9	101.6	+33.3
41316	UPPER	60	30	1.550	0.248	0.1770	289.5	138.6	104.3	+34.3
	LOWER	60	30	1.638	0.271	0.2150	305.7	139.6	104.2	+35.4
41348	UPPER	60	30	1.244	0.167	0.0779	234.0	134.3	105.6	+28.7
	LOWER	60	30	1.207	0.158	0.0692	227.2	133.7	105.6	+28.1

NOTES:

1. PANEL PROPERTIES
   - Z = DISTANCE FROM NEUTRAL AXIS TO EXTREME FIBER, in.
   - A = CROSS-SECTION AREA PER UNIT WIDTH, in.^{2}/in.
   - I = MOMENT OF INERTIA PER UNIT WIDTH, in.^{4}/in.
2. NATURAL FREQUENCY PER FIGURE 14-17B
3. ALLOWABLE SOUND LEVEL PER FIGURE 14-17A
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN = (ALLOWABLE dB/Hz – ENVIRONMENT dB/Hz)

TABLE 14-11. SUMMARY OF WING PANEL SONIC FATIGUE ANALYSIS - TASK I MONOCOQUE ARRANGEMENT - HONEYCOMB SANDWICH CONCEPT

POINT DESIGN REGION	WING SURFACE	SPACING		PANEL PROPERTIES(1)				NATURAL(2) FREQUENCY f, (Hz)	SPECTRUM LEVEL(3)(4) (dB/Hz)			MINIMUM(5) SONIC FATIGUE MARGIN (dB/Hz)
		a (in.)	b (in.)	t_1 (in.)	t_2 (in.)	t_e (in.)	h (in.)		FACE ALLOW.	EDGE ALLOW.	ENVIR.	
40322	UPPER	20	130	0.015	0.015	0.066	0.591	165.7	131.4	131.7	90.6	+40.8
	LOWER	20	130	0.011	0.020	0.078	0.796	230.6	132.2	132.2	90.2	+42.0
40236	UPPER	20	60	0.047	0.049	0.144	0.675	304.6	139.7	140.5	100.2	+39.5
	LOWER	20	60	0.092	0.027	0.119	0.298	154.6	139.4	141.6	101.3	+38.1
40536	UPPER	20	60	0.053	0.052	0.158	0.732	325.6	140.4	141.3	100.6	+39.8
	LOWER	20	60	0.076	0.061	0.103	0.153	96.3	138.9	142.5	102.6	+36.3
41036	UPPER	20	60	0.033	0.033	0.116	0.627	278.6	137.1	137.9	101.4	+35.7
	LOWER	20	60	0.047	0.036	0.083	0.175	93.8	136.0	139.2	103.2	+32.8
41316	UPPER	20	60	0.052	0.053	0.158	0.617	283.5	140.1	142.1	104.3	+35.8
	LOWER	20	60	0.070	0.062	0.066	0.060	54.9	136.8	140.8	107.0	+29.8
41348	UPPER	20	60	0.035	0.038	0.128	0.646	282.9	137.8	139.1	105.3	+32.5
	LOWER	20	60	0.047	0.044	0.068	0.130	75.8	136.0	137.9	108.0	+28.0

NOTES:
1. PANEL PROPERTIES
   - t_1 = INTERIOR FACE SHEET THICKNESS, in.
   - t_2 = EXTERIOR FACE SHEET THICKNESS, in.
   - t_e = EDGE THICKNESS, in.
   - h = PANEL HEIGHT, in.
2. NATURAL FREQUENCY PER FIGURE 14-19
3. ALLOWABLE SOUND LEVELS PER FIGURE 14-18
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN: (ALLOWABLE dB/Hz – ENVIRONMENT dB/Hz)

Fuselage Analysis

Sonic fatigue analyses were conducted on the most promising combination of fuselage concepts during the Task I detailed concept analysis. No sonic fatigue analyses were conducted during the initial screening phase of Task I.

The analysis was conducted on the least weight concept for each of the four point design regions. The locations of the point design regions are presented in Figures 14-24, the associated structural concepts for these regions are:

- FS 750 - Zee stiffened concept
- FS 2000, 2500, and 3000 - hat stiffened concept

The corresponding panel dimensions and equivalent thicknesses for these concepts are displayed in Table 14-12. A study was conducted to compare the capability of the entire panel between points of attachment and the skin between stiffeners to resist sonic fatigue. The results of this study, which was conducted at FS 3000, are summarized in Table 14-13 and include the spectrum levels and natural frequencies of both components. With reference to this table, the skin afforded a higher resistance to sonic fatigue than the panels (i.e., skin allowable spectrum levels were approximately 2- to 3-percent higher than the panel values) and as a result the panel allowable spectrum levels were used in all further analysis. Periodically checks were conducted to insure this relationship held for all regions.

A summary of the results of the fuselage sonic fatigue analyses is shown in Table 14-14. These calculation were determined using the same methods and design charts used for the wing orthotropic panel analyses. The panel properties, natural frequencies, and the allowable and applied spectrum levels are displayed on this table. All fuselage regions have a positive sonic fatigue margin with the minimum margin (+16 dB/Hz) occuring on the side panel at FS3000.

In conclusion, positive sonic fatigue margins exist on the Task I fuselage structure at all point design regions and no additional stiffness or associated weight penalty was required to meet the sonic fatigue requirements.

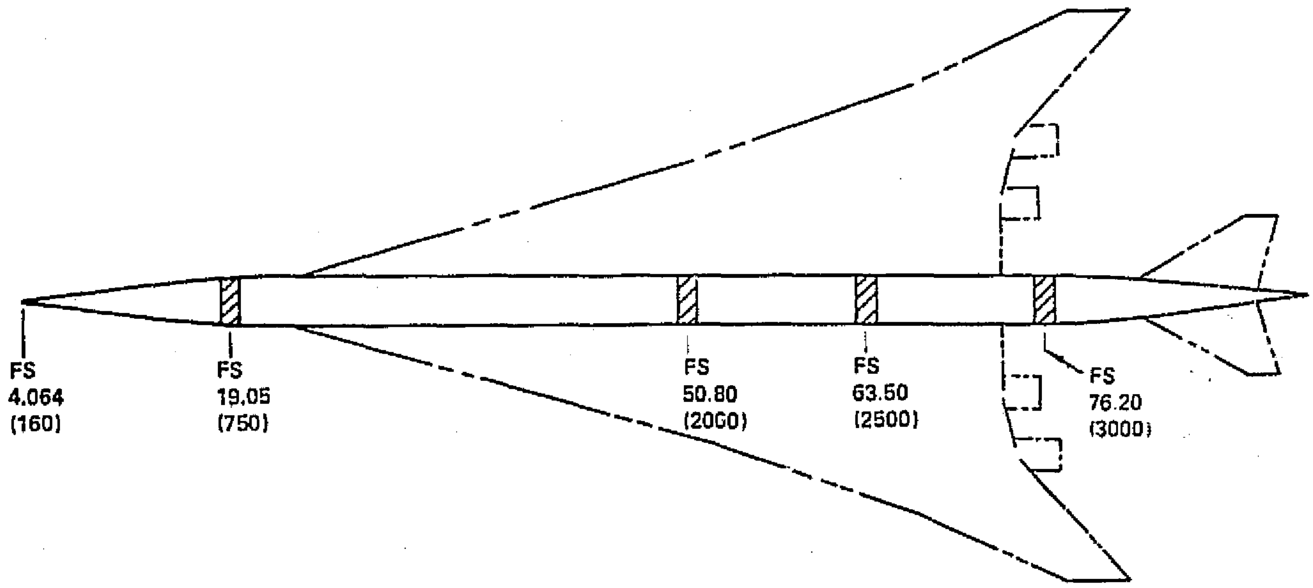


Figure 14-24. Definition of Fuselage Point Design Regions - Task I

TABLE 14-12. FUSELAGE PANEL GEOMETRY - TASK I DETAILED CONCEPT ANALYSIS

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	FUSELAGE PANEL DIMENSION						
			b_s (IN.)	t_s (IN.)	C (IN.)	f (IN.)	h (IN.)	t_{st} (IN.)	$\bar{t}$ (IN.)
FS 750	ZEE-STIFFENED	TOP	4.0	.036	.55	.75	1.00	.036	.056
		SIDE	4.0	.036	.55	.75	1.00	.036	.056
		BOTTOM	4.0	.036	.55	.75	1.00	.036	.056
FS 2000	HAT-STIFFENED	TOP	6.0	.080	1.5	.80	1.25	.070	.145
		SIDE	6.0	.063	1.5	.75	1.25	.040	.099
FS 2500	HAT-STIFFENED	TOP	6.0	.090	1.5	.80	1.25	.090	.174
		SIDE	6.0	.063	1.5	.75	1.25	.050	.109
FS 3000	HAT-STIFFENED	TOP	6.0	.080	1.5	.80	1.25	.070	.145
		SIDE	6.0	.063	1.5	.75	1.25	.040	.099
		BOTTOM	6.0	.090	1.5	.90	1.25	.090	.177

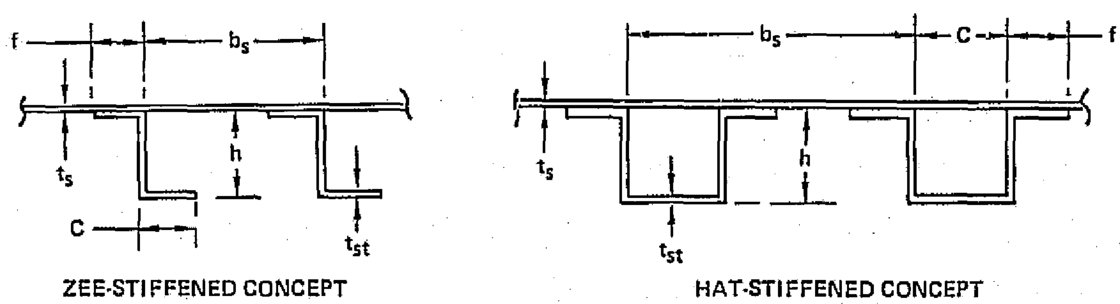
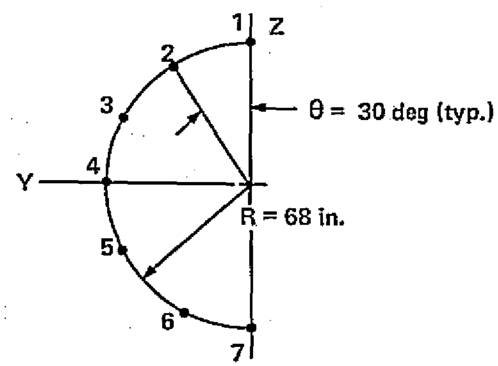


TABLE 14-13. COMPARISON OF FUSELAGE COMPONENT SONIC FATIGUE ALLOWABLES - TASK I

POINT DESIGN REGION	CIRCUMFERENTIAL[1] LOCATION	SKIN VALUES[2] FREQUENCY, f, (Hz)	SKIN VALUES[2] SPECTRUM LEVEL (dB/Hz)	PANEL VALUES[3] FREQUENCY, f, (Hz)	PANEL VALUES[3] SPECTRUM LEVEL (dB/Hz)
FS 3000	1	825.9	137.1	362.7	133.6
	2	706.7	134.9	364.7	132.7
	3	636.0	133.3	357.1	130.8
	4	636.0	133.3	367.4	132.2
	5	706.7	134.9	350.8	131.0
	6	825.9	137.1	362.7	133.6
	7	972.2	139.2	367.4	135.6

NOTES: 1. CIRCUMFERENTIAL LOCATION



2. VALUES OF THE SKIN BETWEEN STIFFENERS
3. VALUES OF THE PANEL, INCLUDES STIFFNESS OF SKIN AND STRINGER.

TABLE 14-14. SUMMARY OF FUSELAGE SONIC FATIGUE ANALYSES, DETAILED CONCEPT ANALYSIS - TASK I

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	SPACING, (in.)		PANEL PROPERTIES(1)			NATURAL(2) FREQUENCY f, (Hz)	SPECTRUM LEVEL(3)(4) (dB/Hz)		SONIC(5) FATIGUE MARGIN (dB/Hz)
			a	b_s	Z (in.)	A (in.^{2}/in.)	I (in.^{4}/in.)		ALLOW.	ENVIR.	
FS 750	ZEE-STIFFENED	TOP	20	4.0	0.848	0.056	0.0063	270.0	121.5	83.5	+38.0
		SIDE	20	4.0	0.848	0.056	0.0063	270.0	121.5	83.5	+38.0
		BOTTOM	20	4.0	0.848	0.056	0.0063	270.0	121.5	83.5	+38.0
FS 2000	HAT-STIFFENED	TOP	20	6.0	0.993	0.144	0.0318	362.7	133.6	91.2	+42.4
		SIDE	20	6.0	1.039	0.099	0.0195	342.0	129.2	91.1	+38.1
FS 2500	HAT-STIFFENED	TOP	20	6.0	0.950	0.161	0.0380	374.4	135.4	99.0	+36.4
		SIDE	20	6.0	1.003	0.108	0.0231	357.1	130.8	99.2	+31.6
FS 3000	HAT-STIFFENED	TOP	20	6.0	0.993	0.144	0.0318	362.7	133.6	116.0	+17.6
		SIDE	20	6.0	0.973	0.117	0.0265	367.4	132.2	116.0	+16.2
		BOTTOM	20	6.0	0.977	0.174	0.0395	367.4	135.6	116.0	+19.6

NOTES:
1. PANEL PROPERTIES
   Z = DISTANCE FROM NEUTRAL AXIS TO EXTREME FIBER, in.
   A = CROSS-SECTION AREA PER UNIT WIDTH, in.^{2}/in.
   I = MOMENT OF INERTIA PER UNIT WIDTH, in.^{4}/in.
2. NATURAL FREQUENCY PER FIGURE 14-17B
3. ALLOWABLE SOUND LEVEL PER FIGURE 14-17A
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN = (ALLOWABLE dB/Hz — ENVIRONMENT dB/Hz)

SONIC FATIGUE ANALYSIS - TASK II

For the Task II Detail Engineering Studies, the final wing and fuselage structural arrangement was subjected to a detail sonic satigue analysis. Similar to the Task I effort, this analysis was restricted to evaluating the surface panels capability only.

Wing Analysis

Analyses were conducted on the upper and lower surface panels at the six wing point design regions. The wing point design locations previously displayed in Figure 14-23 are also appropriate for the Task II effort.

The Final Design airplane incorporates both the chordwise convex-beaded and the monocoque honeycomb sandwich surface panel designs. With reference to Figure 14-23, the convex-beaded concept is utilized at point design regions 40322, 40236, and 40536 and the honeycomb sandwich concept at regions 41036, 41316, and 41348.

Chordwise Arrangement - The surface panel geometry for the convex-beaded concept is presented in Table 14-15 and reflects the results of the strength analysis. These data include the minimum weight panel proportions (rib/spar spacing), cross-sectional dimensions, panel weight data, and the critical design condition used for the strength analysis.

The convex-beaded panels were analyzed using the previously described methods and the design charts outlined in Figure 14-17. Table 14-16 contains a summary of the analysis results, which include the panel properties and the applied and allowable sonic spectrum levels. A minimum sonic fatigue margin of +12.0 dB/Hz is indicated for the lower surface panel at point design region 40536. Conversely, the maximum margin occurs on the lower panel at region 40322, +24.2 dB/Hz.

Monocoque Arrangement - The honeycomb sandwich panels were analyzed at regions 41036, 41316, and 41348. The panel geometry associated with these regions are presented in Table 14-17. The geometry associated with regions 41316 and 41348 reflect the stiffness required to meet the flutter criteria; whereas, regi[illegible] 036 is strength designed.

The design charts presented in Figure 14-18 and 14-9 were used to define the panel face sheet allowable and natural frequency respectively. The panel edge allowable was defined from the design chart in Figure 83 of Reference 4. The applied

TABLE 14-15. WING PANEL GEOMETRY - TASK IIB, CONVEX BEADED PANELS

DESIGN DATA	POINT DESIGN REGIONS					
	40322		40236		40536	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
RIB	60.0	60.0	60.0	60.0	60.0	60.0
SPAR	22.7	22.7	21.2	21.2	21.2	21.2
DIMENSIONS						
t_L, in.	.013	.015	.015	.020	.023	.019
t_u, in.	.015	.020	.015	.020	.026	.020
R_L, in.	.80	1.00	.80	1.00	.90	.70
θ, degrees	87	87	87	87	87	87
b, in.	.75	.75	.75	.75	.75	.75
pitch, in.	2.35	2.75	2.35	2.75	2.55	2.15
WEIGHT DATA						
$\bar{t}$, in.	.033	.041	.036	.048	.058	.046
W, lb./sq.ft.	.760	.945	.829	1.11	1.34	1.05
CRITICAL DESIGN COND.	12	20	16	16	12	12

DIMENSIONS:

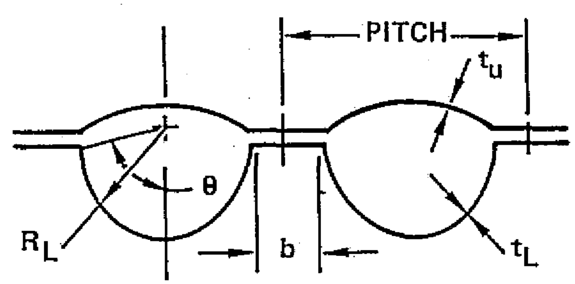


TABLE 14-16. SUMMARY OF WING CONVEX-BEADED PANEL ANALYSES - TASK IIB

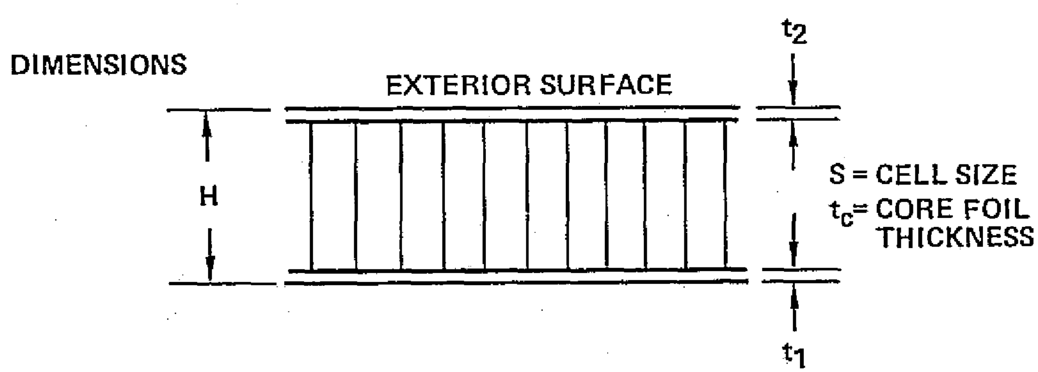
POINT DESIGN REGION	WING SURFACE	SPACING (in.)		PANEL PROPERTIES(1)			NATURAL(2) FREQ. (f) (Hz)	SPECTRUM LEVEL(3)(4) (dB/Hz)		SONIC(5) FATIGUE MARGIN (dB/Hz)
		a	b	Z (in.)	A (in.^{2}/in.)	I (in.^{4}/in.)		ALLOW.	ENVIR.	
40322	UPPER	22.7	60.0	0.594	0.033	0.00133	120.4	110.0	91.0	+19.0
	LOWER	22.7	60.0	0.751	0.041	0.00226	140.8	115.0	90.8	+24.2
40236	UPPER	21.2	60.0	0.578	0.036	0.00147	138.9	114.0	101.6	+12.4
	LOWER	21.2	60.0	0.710	0.048	0.00275	164.5	118.0	101.2	+16.8
40536	UPPER	21.2	60.0	0.660	0.058	0.00281	151.3	120.0	101.8	+18.2
	LOWER	21.2	60.0	0.516	0.046	0.00153	125.4	114.0	102.0	+12.0

NOTES:

1. PANEL PROPERTIES
   - Z = DISTANCE FROM NEUTRAL AXIS TO EXTREME FIBER, in.
   - A = CROSS-SECTION AREA PER UNIT WIDTH, in.^{2}/in.
   - I = MOMENT OF INERTIA PER UNIT WIDTH, in.^{4}/in.
2. NATURAL FREQUENCY PER FIGURE 14-17B
3. ALLOWABLE SOUND LEVEL PER FIGURE 14-17A
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN = (ALLOWABLE dB/Hz — ENVIRONMENT dB/Hz)

TABLE 14-17. WING PANEL GEOMETRY - TASK IIB, HONEYCOMB SANDWICH PANELS

DESIGN DATA	POINT DESIGN REGIONS					
	41036		41316		41348	
	UPPER	LOWER	UPPER	LOWER	UPPER	LOWER
SPACING, in.						
RIB	60.0	60.0	40.0	40.0	40.0	40.0
SPAR	21.2	21.2	40.0	40.0	30.0	30.0
DIMENSIONS						
H, in.	.642	.202	1.00	.500	1.00	.500
t_1, in.	.026	.023	.062	.075	.068	.068
t_2, in.	.018	.028	.062	.075	.068	.068
t_c, in.	.002	.002	.002	.002	.002	.002
S, in.	.275	.500	.500	.500	.500	.500
WEIGHT DATA						
$\bar{t}$, in.	.052	.052	.131	.153	.143	.139
W, lb./sq.ft.	1.20	1.20	3.02	3.52	3.29	3.20
CRITICAL DESIGN COND.	12	12	FLUTTER	FLUTTER	FLUTTER	FLUTTER



noise levels are shown in Figure 14-21. A summary of the results is displayed in Table 14-18 with positive sonic fatigue margins indicated for all regions.

A minimum margin of +30 dB/Hz occurs on the lower surface at point design region 41036. No adjustment in panel proportions was required to meet the acoustic criteria; hence, no weight penalties were required.

Fuselage Analysis

Sonic fatigue analyses were conducted on the fuselage concepts of the final design airplane at the four fuselage point design regions, Figure 14-25 displays the locations of the fuselage point design regions.

The structural concepts for each fuselage region were identical to those specified for the Task I fuselage and are repeated here for completeness; they are: zee-stiffened concept at FS 900 and the hat-stiffened concept at regions FS 1910, FS 2525, and FS 2900.

The panel geometry associated with the above concepts is presented in Table 14-19 which includes the geometry for all circumferential locations, from the uppermost panel (top) to the lowest panel (bottom). The panel identification system corresponds to that used for the NASTRAN model element identification and is shown in Figure 14-26. For ease in reporting, only the upper, side and bottom panels are presented. With respect to Figure 14-26, these panels are identified by the last two digits of the NASTRAN element number: 01, 06, and 09 respectively.

A summary of the results of the fuselage analysis is presented in Table 14-20. This analysis was conducted using the design charts displayed in Figure 14-17 with the pertinent section properties in the above Table. The resulting natural frequencies and allowable spectrum levels obtained from these charts were compared to the environmental levels determined from Figure 14-22. With reference to Table 14-20, positive sonic fatigue margins are indicated with a minimum margin of +9.8 dB/Hz occuring on the side panel at FS 2900.

TABLE 14-18. SUMMARY OF WING HONEYCOMB SANDWICH PANEL ANALYSES - TASK IIB

POINT DESIGN REGION	WING SURFACE	SPACING		PANEL PROPERTIES[1]				NATURAL[2] FREQUENCY f, (Hz)	SPECTRUM LEVEL[3][4] (dB/Hz)			MINIMUM[5] SONIC FATIGUE MARGIN (dB/Hz)
		a (in.)	b (in.)	t_1 (in.)	t_2 (in.)	t_e (in.)	h (in.)		FACE SHT. ALLOW.	EDGE ALLOW.	ENVIR.	
41036	UPPER	21.2	60.0	0.026	0.018	0.091	0.598	234	137.9	139.3	101.6	+36.3
	LOWER	21.2	60.0	0.023	0.028	0.088	0.151	66	136.2	144.8	106.0	+30.2
41316	UPPER	40.0	40.0	0.062	0.062	0.167	0.876	175	146.6	146.0	105.0	+41.6
	LOWER	40.0	40.0	0.075	0.075	0.194	0.350	79	146.0	151.9	106.8	+39.2
41348	UPPER	30.0	40.0	0.068	0.068	0.179	0.864	264	146.7	146.3	105.4	+41.3
	LOWER	30.0	40.0	0.068	0.068	0.179	0.364	122	145.0	150.0	106.6	+38.4

NOTES:
1. PANEL PROPERTIES
   - t_1 = INTERIOR FACE SHEET THICKNESS, in.
   - t_2 = EXTERIOR FACE SHEET THICKNESS, in.
   - t_e = EDGE THICKNESS, in.
   - h = CORE HEIGHT, in.
2. NATURAL FREQUENCY PER FIGURE 14-19
3. ALLOWABLE SOUND LEVELS PER FIGURE 14-18
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN: (ALLOWABLE dB/Hz) – ENVIRONMENT dB/Hz)

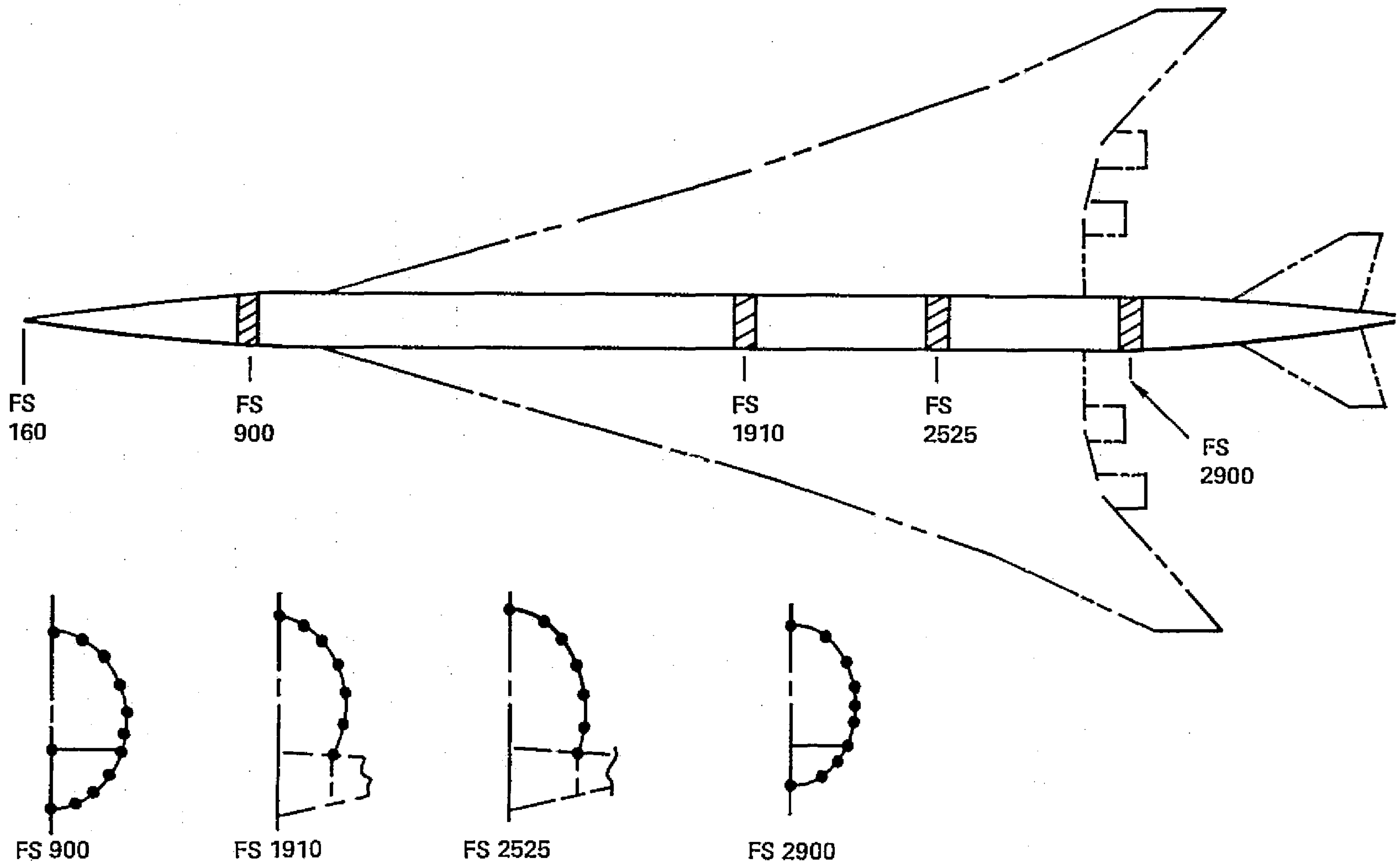
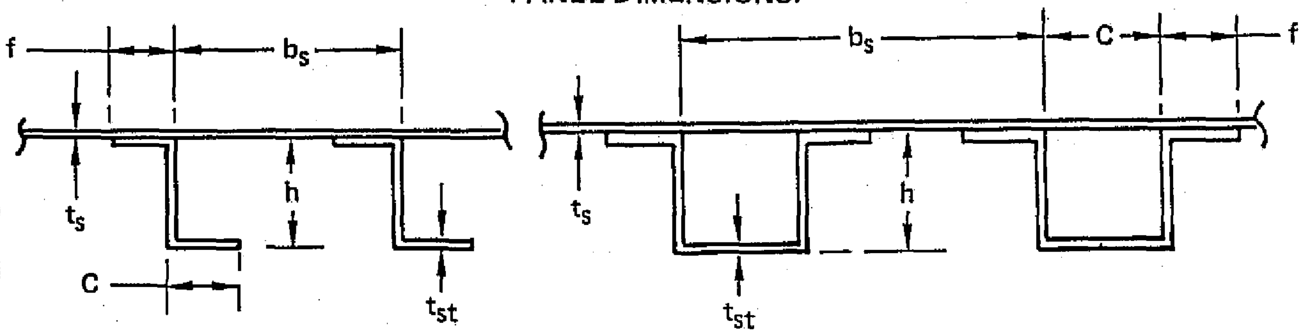


Figure 14-25. Definition of Fuselage Point Design Regions - Task IIB

TABLE 14-19. FUSELAGE PANEL GEOMETRY - TASK IIB

POINT DESIGN REGION	PANEL CONCEPT	CIRCUMF. LOCATION	FUSELAGE PANEL DIMENSIONS						
			b_s (in.)	t_s (in.)	C (in.)	f (in.)	h (in.)	t_{st} (in.)	$\bar{t}$ (in.)
FS 900	ZEE-STIFFENED	233301-233307	4.0	.036	.55	0.75	1.00	.036	.056
FS 1910	HAT-STIFFENED	234101	6.0	.07	1.5	0.80	1.25	.06	.129
		234102	6.0	.06	1.5	0.80	1.25	.05	.109
		234103	6.0	.04	1.5	0.80	1.25	.04	.079
		234104	6.0	.04	1.5	0.80	1.25	.03	.069
		234105	6.0	.05	1.5	0.80	1.25	.05	.099
		234106	6.0	.06	1.5	0.80	1.25	.06	.119
FS 2525	HAT-STIFFENED	234801	6.0	.07	1.5	0.80	1.25	.08	.149
		234802	6.0	.06	1.5	0.80	1.25	.06	.119
		234803	6.0	.05	1.5	0.80	1.25	.05	.099
		234804	6.0	.04	1.5	0.80	1.25	.03	.069
		234805	6.0	.04	1.5	0.80	1.25	.03	.069
		234806	6.0	.04	1.5	0.80	1.25	.04	.079
FS 2900	HAT-STIFFENED	235101	6.0	.07	1.5	0.80	1.25	.07	.139
		235102	6.0	.05	1.5	0.80	1.25	.06	.109
		235103	6.0	.05	1.5	0.80	1.25	.04	.089
		235104	6.0	.04	1.5	0.80	1.25	.03	.069
		235105	6.0	.04	1.5	0.80	1.25	.03	.069
		235106	6.0	.04	1.5	0.80	1.25	.03	.069
		235107	6.0	.05	1.5	0.80	1.25	.04	.089
		235108	6.0	.05	1.5	0.80	1.25	.06	.109
		235109	6.0	.07	1.5	0.80	1.25	.08	.149

PANEL DIMENSIONS:



ZEE-STIFFENED CONCEPT

HAT-STIFFENED CONCEPT

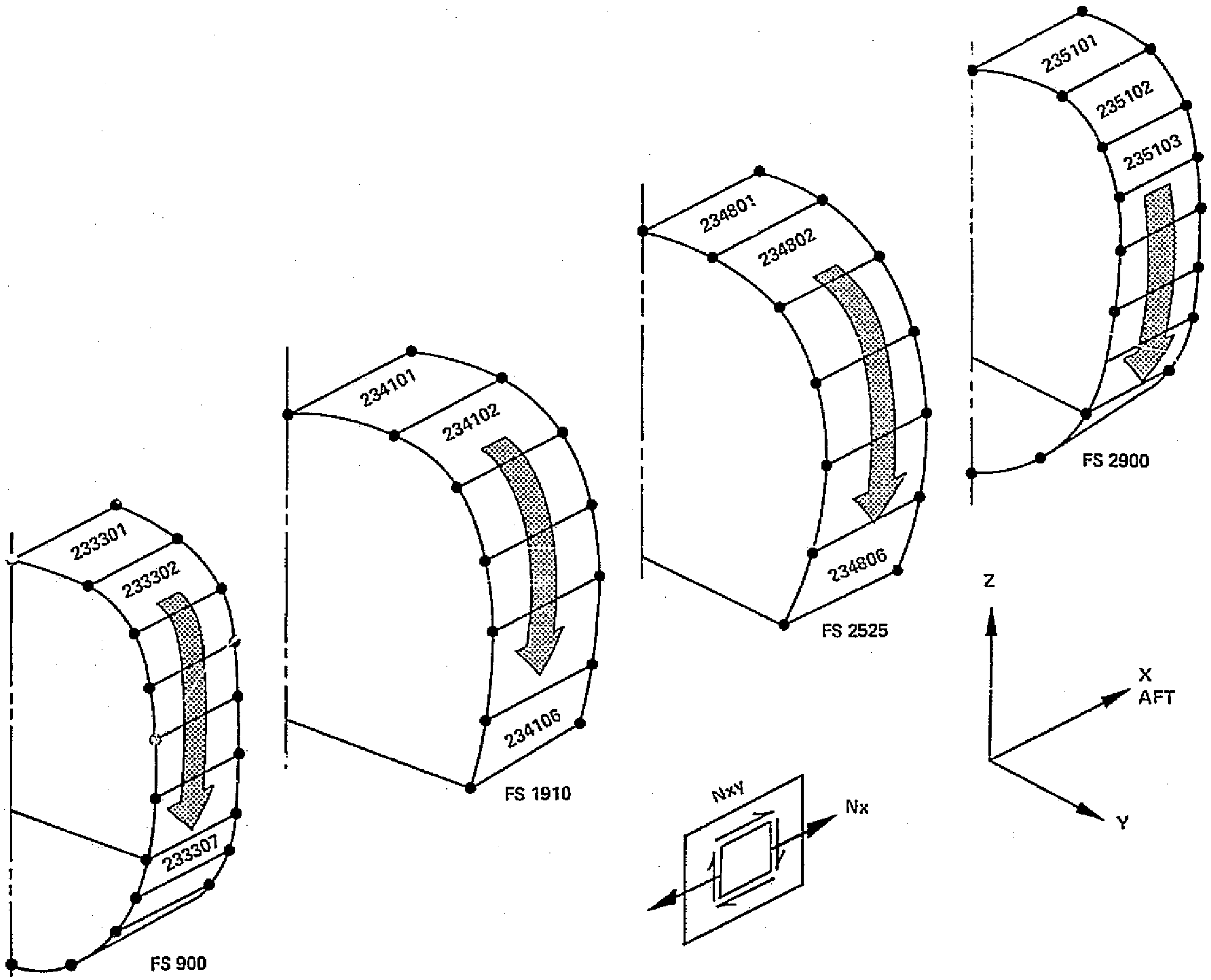


Figure 14-26. Fuselage Panel Identification - Task IIB

TABLE 14-20. SUMMARY OF FUSELAGE PANELS SONIC FATIGUE ANALYSES - TASK IIB

POINT DESIGN REGION	PANEL CONCEPT	LOCATION	SPACING, (in.)		PANEL PROPERTIES[1]			NATURAL[2] FREQUENCY f, (Hz)	SPECTRUM LEVEL[3][4] (dB/Hz)		SONIC[5] FATIGUE MARGIN (dB/Hz)
			a	b_s	Z (in.)	A (in.^{2}/in.)	I (in.^{4}/in.)		ALLOW.	ENVIR.	
FS 900	ZEE-STIFFENED	TOP	20.9	4.0	0.848	0.056	0.0063	245.0	121.0	84.0	+37.0
		SIDE	20.9	4.0	0.848	0.056	0.0063	245.0	121.0	84.0	+37.0
		BOTTOM	20.9	4.0	0.848	0.056	0.0063	245.0	121.0	84.0	+37.0
FS 1910	HAT-STIFFENED	TOP	22.7	6.0	0.994	0.129	0.0281	255.0	131.0	92.0	+39.0
		SIDE	22.7	6.0	0.966	0.119	0.0270	255.0	130.5	92.0	+38.5
FS 2525	HAT-STIFFENED	TOP	21.2	6.0	0.946	0.149	0.0351	350.0	134.0	99.4	+34.6
		SIDE	21.2	6.0	0.948	0.079	0.0182	330.0	128.0	99.6	+28.4
FS 2900	HAT-STIFFENED	TOP	21.0	6.0	0.968	0.139	0.0317	340.0	132.0	116.2	+15.8
		SIDE	21.0	6.0	1.000	0.069	0.0142	300.0	126.0	116.2	+9.8
		BOTTOM	21.0	6.0	0.946	0.149	0.0351	350.0	134.0	116.1	+17.9

NOTES:
1. PANEL PROPERTIES
   Z = DISTANCE FROM NEUTRAL AXIS TO EXTREME FIBER, in.
   A = CROSS-SECTION AREA PER UNIT WIDTH, in.^{2}/in.
   I = MOMENT OF INERTIA PER UNIT WIDTH, in.^{4}/in.
2. NATURAL FREQUENCY PER FIGURE 14-17B
3. ALLOWABLE SOUND LEVEL PER FIGURE 14-17A
4. APPLIED SOUND LEVEL (ENVIRONMENT) PER FIGURE 14-21
5. SONIC FATIGUE MARGIN = (ALLOWABLE dB/Hz – ENVIRONMENT dB/Hz)

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2. General Electric Oral Review, Advanced Supersonic Propulsion System Technology Study. Contract NAS3-16950, Modification 3. June 1974
3. Lockheed-California Company: Structural Life-Assurance Manual, SLM No. 5, Sonic Fatigue Prevention.
4. Ballentine, J.R., Et Al.: Refinement of Sonic Fatigue Structural Design Criteria. AFFDL-TR-67-156, January 1968.

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